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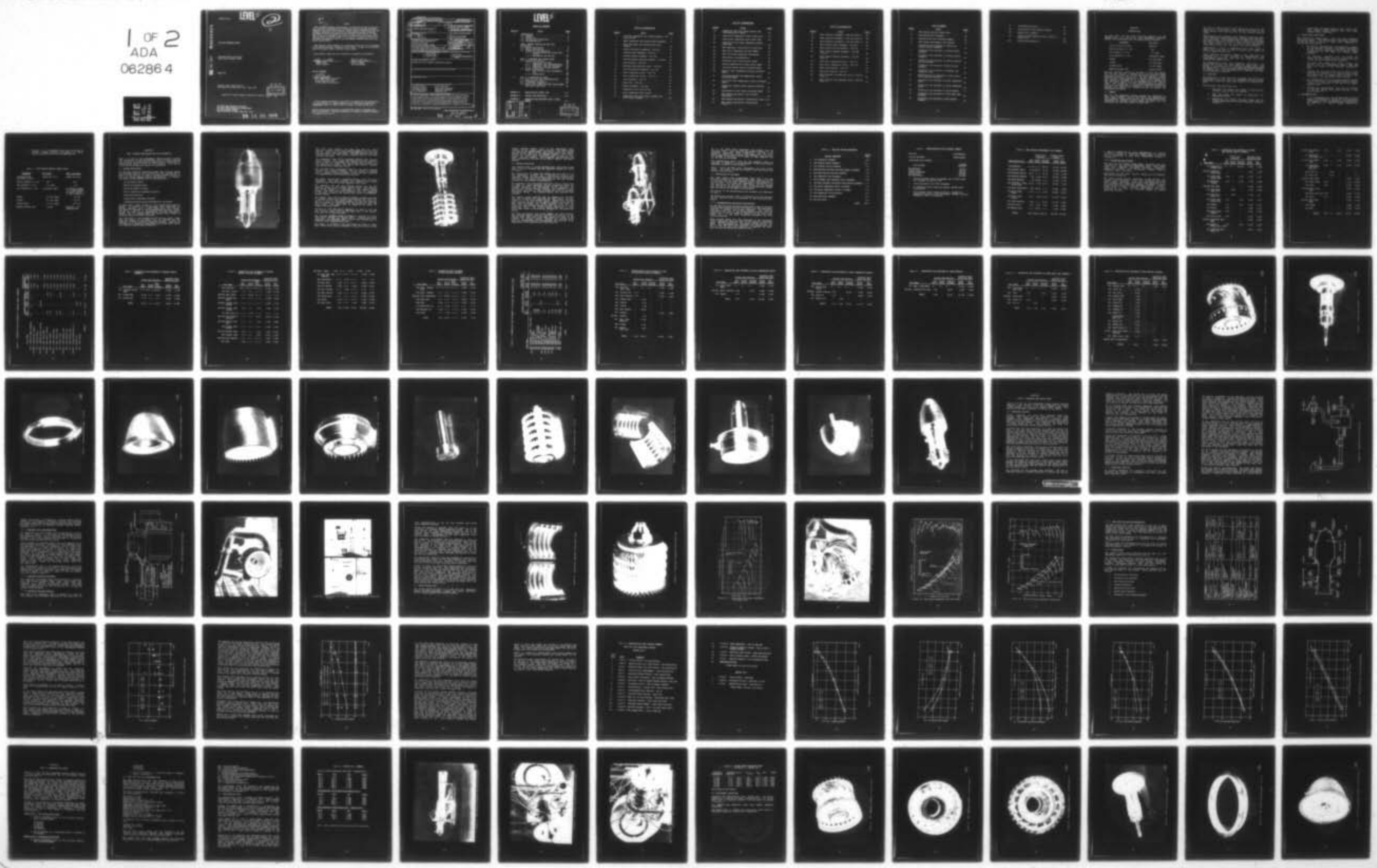

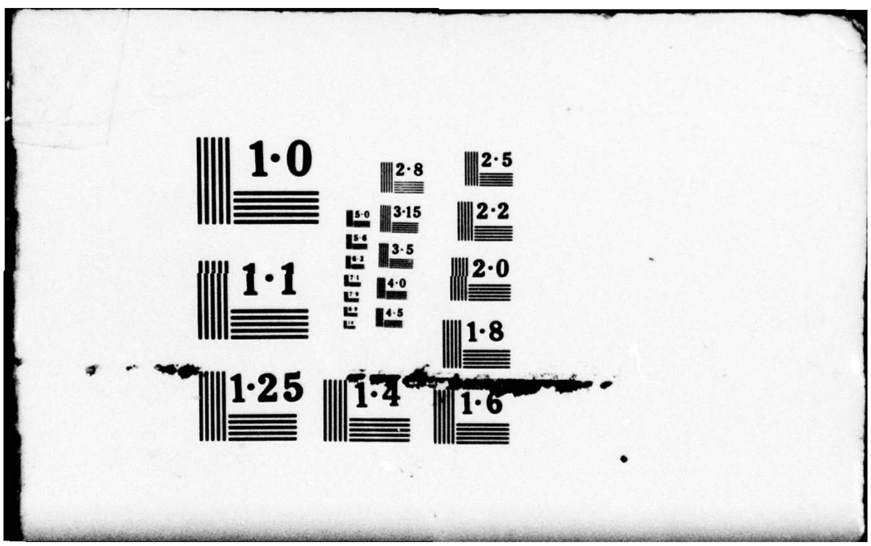
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LEVEL II

AFAPL-TR-78-33

LOW COST EXPENDABLE ENGINE

Williams Research Corporation
Walled Lake, Michigan 48088

MARCH 1978

Technical Report AFAPL-TR-78-33
Final Report for Period April 1976 - March 1978

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AIR FORCE AERO PROPULSION LABORATORY
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This technical report has been reviewed and is approved for publication.

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GAS TURBINE ENGINES	SHEET METAL COMPONENTS
AXIAL COMPRESSORS	COMPRESSOR TEST

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

A low cost expendable turbojet engine in the 200 pound thrust class was fabricated and tested. The design, manufacturing, and inspection concepts of the program resulted in the achievement of a projected engine cost of $2883 each in lots of 1000 engines in terms of 1975 economics. Problems solved during the compressor rig testing and engine tune-up testing are discussed. The results of the engine demonstration testing both at sea level static conditions and under a simulated Mn 0.7 condition are presented.

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LEVEL II

TABLE OF CONTENTS

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS

LIST OF TABLES

SECTION 1

INTRODUCTION

In early 1976, the Air Force solicited proposals for the demonstration of a truly low cost expendable propulsion system. The stated goals for this engine were as follows:

PARAMETER	RFP GOALS
Cost (in lots of 1000)	\$2000.00
Thrust (@ Mach 0.7, SL)	200
SFC (@ Mach 0.7, SL)	3.0 (max)
Life (@ Mach 0.7, SL)	30 Minutes (min)
Weight	100 lbs (max)
Volume	3.0 ft^3 (max)
Frontal Area	1.2 ft^2 (max)
Engine Storage Life	5 years (min)

Williams Research Corporation elected to offer its concept for low cost design, a concept which was initiated about three years earlier and was actively being applied to engine hardware fabrication and tests. The WRC in-house activity was, however, concentrating on a thrust level higher than that desired by the USAF, so all designs were scaled down to the 200 pound thrust size and completed prior to contract award. This design and a detailed processing description of each part was submitted in response to the RFP and WRC was awarded USAF Contract No. F33615-76-C-2123 to conduct the demonstration program.

1.1 SUMMARY

The Low Cost Expendable Engine program was structured in three task elements; engine fabrication and manufacturing cost estimate, preparatory testing, and demonstration tests. Initiation of program activity began in April 1976 and concluded with this final report in March 1978.

Two sets of engine hardware were fabricated along with test spare parts as necessary. Both engines were used in the demonstration tests and both are in good usable condition today.

Manufacturing cost estimates were completed using the realistic experience gained during the tooling tryout and fabrication processes. An estimated price of $2883 (FY 75 dollars) per engine is believed accurate within 30 percent and achieveable when ordered in lots of 1000 to the specification requirements of one start, one speed, and a one-way mission.

Approximately 8 hours of preparatory tests were logged on engine S/N 1 and about 2 hours on engine S/N 2 before the official demonstration tests.

Areas requiring development emphasis were identified and corrected only if they impacted the demonstration test requirements, otherwise, they were set aside for future advanced development effort.

Two demonstration tests were conducted: The first, a 30 minute test at plus 175 pound thrust on engine S/N 1; and second, a 30 minute test at a corrected maximum engine speed on engine S/N 2. Both tests were completed without incident. One engine, S/N 1, was disassembled for inspection and found to be in excellent condition.

1.2 OBJECTIVE

The objective of this report is to present the results and supporting data of the Low Cost Expendable Engine program, successfully completed under USAF Contract No. F33615-76-C-2123.

The individual task objectives were:

1. Fabricate the engine and conduct a detailed cost estimate accurate within ±30 percent.

2. Test and adjust the engine in preparation for demonstration test.

3. Demonstrate the engine SFC and thrust for 30 minutes continuously at the specified test condition of Mach 0.7.

4. Verify that the engine goals of cost, thrust, SFC, life, weight, volume, frontal area, and engine storage life are achievable.

1.3 RESULTS AND CONCLUSIONS

WRC successfully demonstrated a truly low cost expendable propulsion system. The demonstrated achievements relative to the RFP goals are shown on Table 1 and summarized in the following conclusions:

a. The design, manufacturing, and inspection concepts of this program resulted in successful achievement of the low cost objectives and constitute a major breakthrough in the small gas turbine engine business.

b. The individual component costs are within the target and show engine costs to be $2883.00* each in lots of 1000 in terms of 1975 economics.

c. The Model WR33 engine SFC, weight, volume, and frontal area are substantally better than the goals set forth in the Contract F33615-76-C-2123.

d. Although the 200 pound thrust goal at Mach 0.7 sea level was not achieved, it is believed that further development work to reduce burner pressure drop would result in a significant thrust increase.

e. The demonstrated life of the engine greatly exceeds the 30 minute goal. Both engines showed no deterioration in performance at the end of the demonstration test.

f. Although not demonstrated, there are no features in the WR33 engine which would preclude a five year storage life.

1.4 RECOMMENDATIONS

a. It is recommended that the WRC Model WR33 turbojet engine be considered as having met the contractual requirements of cost, performance and producibility as specified in Contract Number F33615-76-C-2123.

b. Further, it is recommended that this technology be pursued through advanced development to assure timely availability for military applications.

TABLE 1. WR33 ENGINE RESULTS VERSUS GOALS

PARAMETER	RFP GOALS	WR33 (ACHIEVED)
Estimated Cost (in lots of 1000)	\$2000 (FY 75 \$)	\$2883 (FY 75 \$)
Thrust (@ Mach 0.7, SL)	200	174
SFC (@ Mach 0.7, SL)	3.0 (Max)	1.72
Life (@ Mach 0.7, SL)	30 minutes (Min)	30 minutes @ Mach 0.7 & Rated Speed (9+ hours engine 1, 3+ hours engine 2)
Weight	100 lbs (Max)	45 lbs
Volume	3.0 ft^3 (Max)	0.8 ft^3
Frontal Area	1.2 ft^2 (Max)	0.57 ft^2
Engine Storage Life	5 years	Feasible - not demonstrated

SECTION 2

TASK I ENGINE FABRICATION AND COST ESTIMATION

Task I of the Low Cost Expendable Engine program involved the fabrication of two test engines, one for tune-up testing and a second for demonstration testing and the estimation of manufacturing costs in lots of 1000 engines. This section of the report presents the results of this effort.

2.1 ENGINE DESCRIPTION

The Williams Research Corporation Model WR33 turbojet engine was designed prior to the initiation of this program and is based on the results of years of company funded efforts on low cost turbine engine studies and analysis. In essence, the design concept embodies the following features.

Low peripheral speed

Modest performance goals

Minimized number of parts

Use of commercial standards and parts

Minimized machining

Single piece compressor castings

Integrated manufacturing and inspection techniques

The Model WR33, shown in Figure 1, is a single-spool turbojet engine. It consists of a six-stage axial compressor, an annular combustor, and a single stage axial flow turbine. The shaft is mounted on two grease-packed bearings, eliminating the need for a pressurized, filtered lubrication system. Air is impinged upon the turbine rotor blades to crank the engine for start initiation.

Fuel delivered to the forward end of the main shaft, flows down the center of the shaft, and is injected into the combustor through oil-burner atomizing nozzles. Ignition is achieved by an automotive type spark plug whose power source is a remotely located ignition exciter.

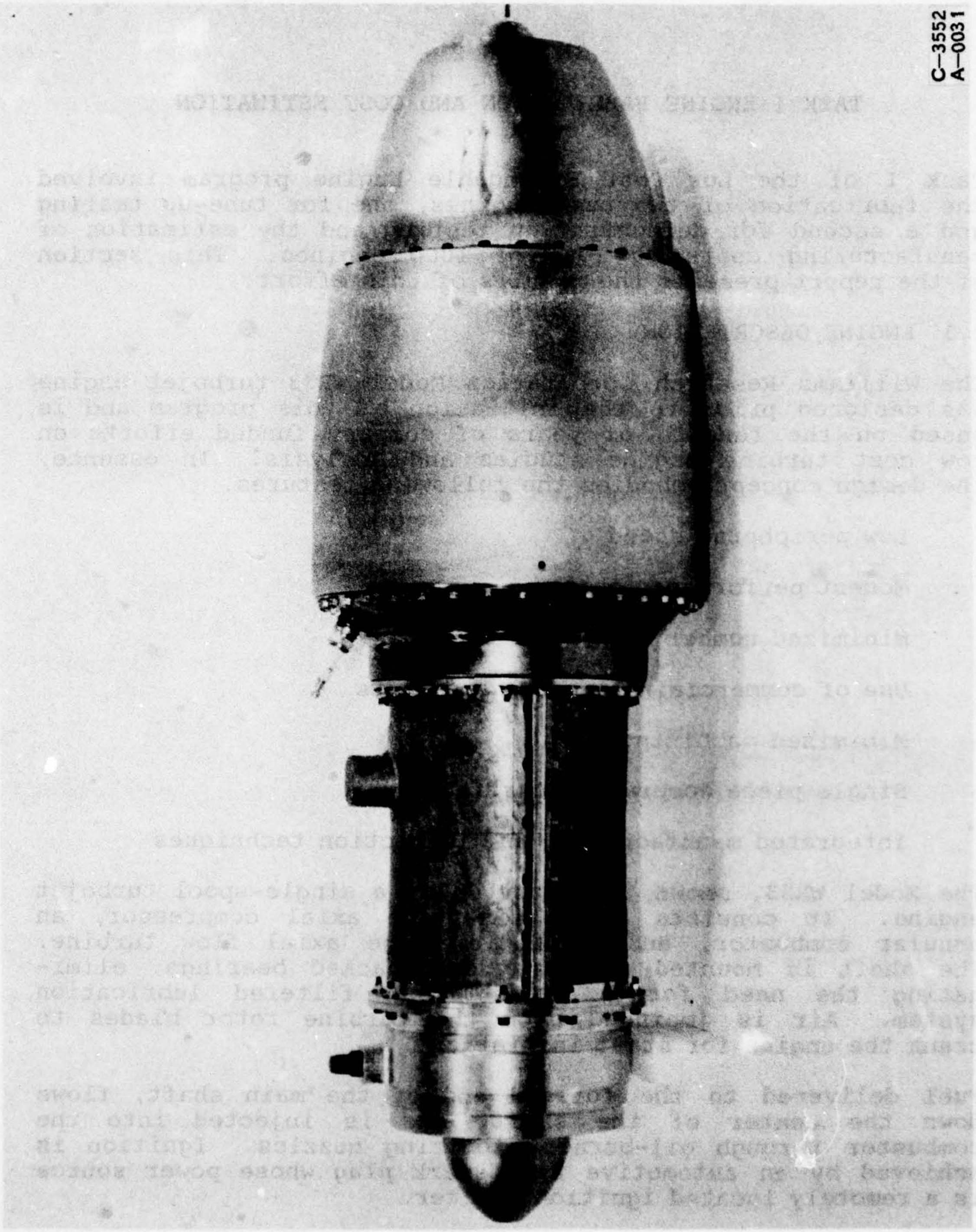

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Figure 1. Low Cost Expendable Gas Turbine Engine - 200 pound Thrust

The inlet case supports the nose cone and the thrust bearing. Struts in the inlet case accommodate the fuel line and the center shaft air vent. The fuel transfer housing, located within the inlet struts, conducts fuel to the shaft and vents the shaft.

The compressor case is an aluminum casting with integral stator blading. Each stator row contains an even number of vanes so that the compressor case may be split axially into two symmetrical halves. The split case permits assembly with the one piece compressor rotor.

The six-stage axial compressor rotor is a one piece aluminum casting with integral blading. The six stages of blading have a constant outside diameter. Flow path area reduction is accomplished by contouring the rotor drum.

The engine incorporates a vaneless diffuser after the axial compressor. The inner diffuser wall supports the forward labyrinth seal carrier and the combustor front cover.

The combustor case and main housing extend from the aft diffuser flange to the turbine shroud flange. The combustor and nozzle assembly is supported at the turbine shroud flange. The inner and outer combustor walls are supported by the nozzle assembly and contain slip joints to accommodate thermal growth.

The exhaust duct and rear bearing support assembly provides a suitably shaped hot exhaust gas passage and final jet nozzle as well as structural support for the rear bearing. The gas passage is established by the concentric inner and outer walls. The inner wall is supported by three "Z" shaped sheet metal vanes.

Cooling air from the axial compressor is used to cool the rear bearing, then flows up the back face of the turbine rotor and out through the nozzle.

The rotating assembly, shown in Figure 2, consists of a main shaft which includes the turbine rotor, a single piece cast compressor, two bearings, and a nut at the forward end to clamp the thrust bearing inner race.

The engine is mounted to the test stand by means of three small sheet metal plates attached to the bolt circle at the

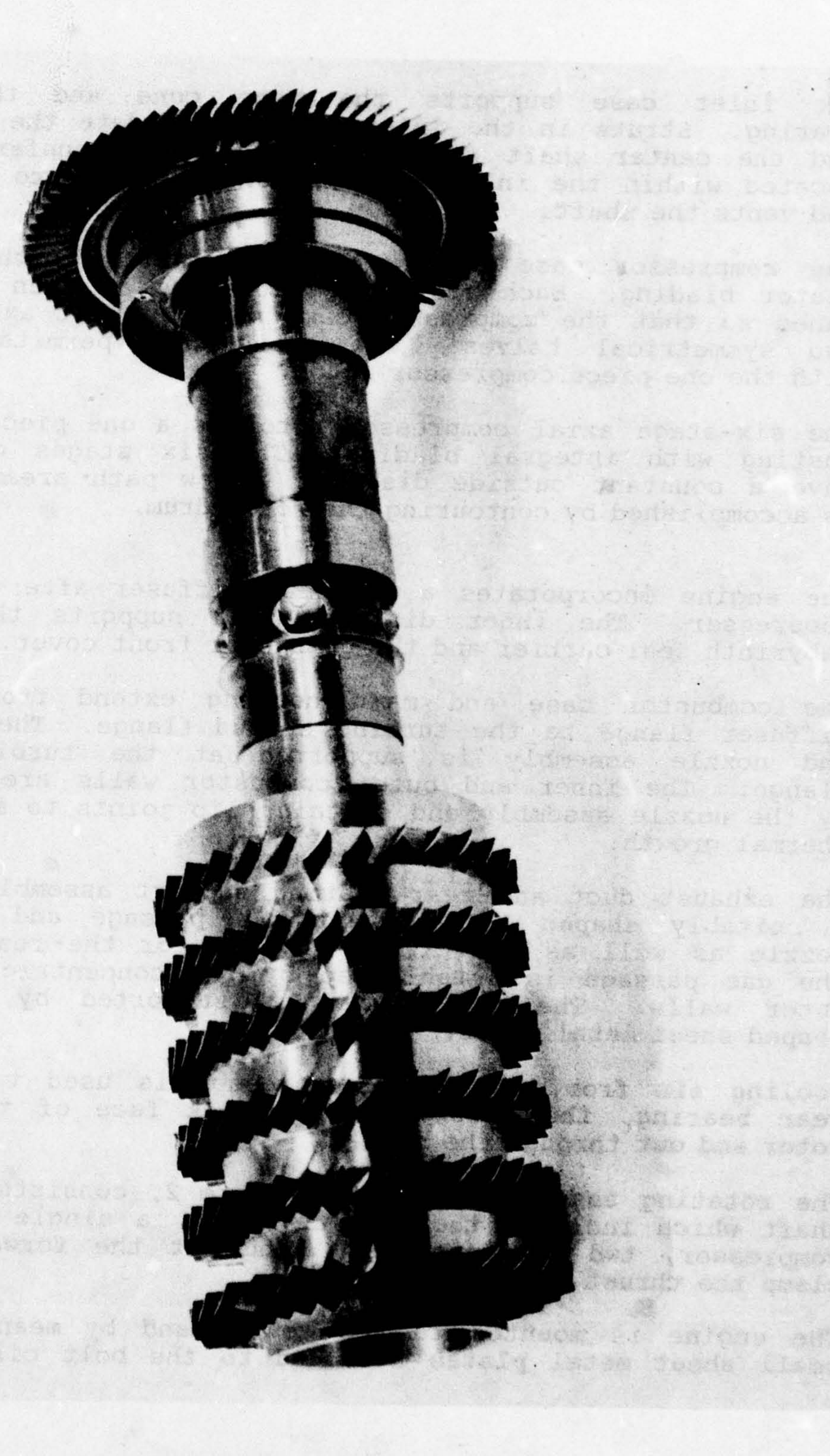

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Figure 2. WR33 Compressor and Turbine Shaft Assembly

larger diffuser flange. This is the approximate axial location of the engine center of gravity. The three plates are positioned approximately 120° apart around the circumference of the engine at its maximum diameter. This configuration will react axial and radial loads as well as roll, pitch, and yaw moments. This mounting configuration has been successfully utilized on the WR2-6, WR24-6, and WR24-7 engines.

2.2 ENGINE FABRICATION

Two complete sets of engine hardware were fabricated during Task I of this program and the completed engines are shown in Figure 3.

The construction of these two engines was such that all of the significant features of a production version were included. Except for the turbine rotor and three potentially cast parts, the engine was fabricated by the same methods proposed for the production configuration.

The compressor rotor and stator utilized a bicast construction in order to insure adequate fatigue properties under the extended operating life requirements of the compressor rig testing and engine tune-up testing. The compressor rotor and stator were cast of C355 aluminum as would the integral production design. The blading however was extruded 6063 aluminum.

The turbine rotors used for the two engines were obtained from WRC inventory and represent the only detail not subjected to a low cost design review. These rotor castings were available as part of the WR19-9 turbofan stock and are made from investment cast IN792 material. The WR33 may retain the same casting process but will use IN 713 LC or an equivalent alloy. The turbine shafts were made of 4340 alloy steel which is the material selected for the production design.

The three potentially cast parts, namely, the inlet housing, rear bearing support housing, and the rear portion of the turbine shaft were made from plate and/or bar stock in the interest of program economy. Tooling for casting these parts is uncomplicated and conventional, but was felt to be unneccesary to the demonstration test objective.

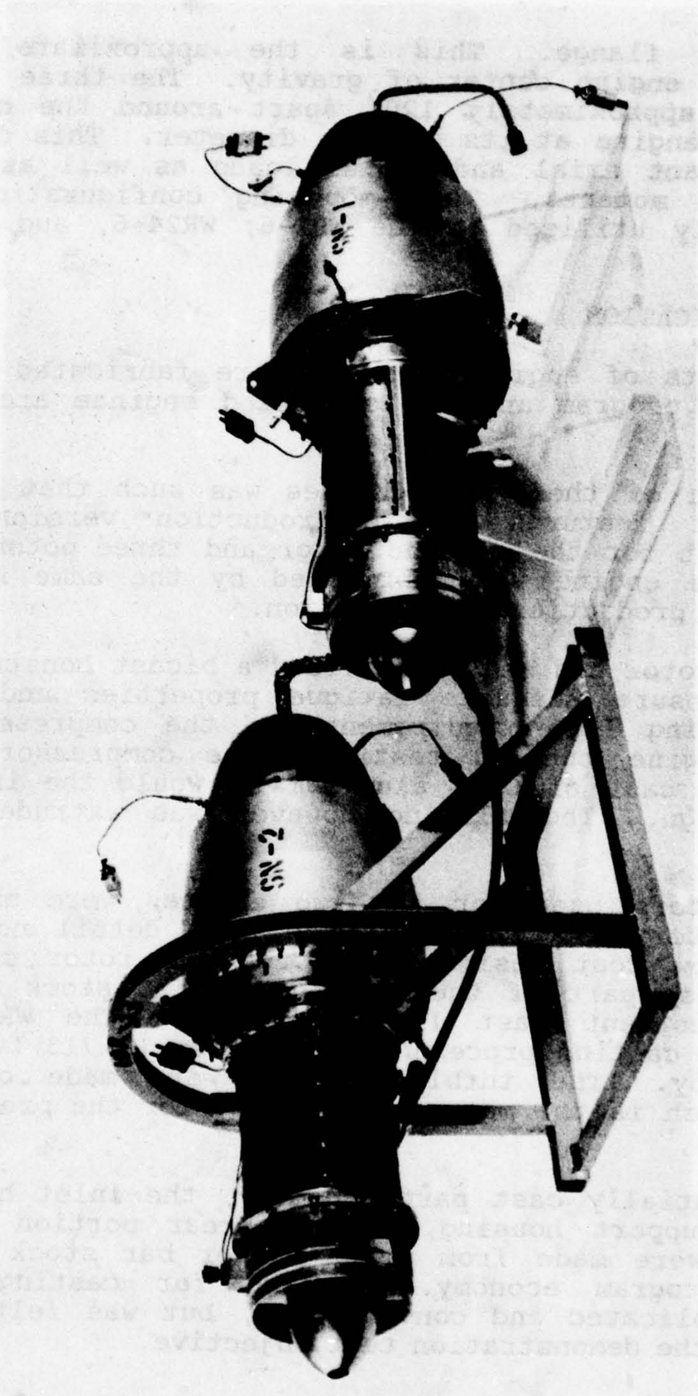


Figure 3. WR33 Prototype and Demonstration Test Engines

All of the sheet metal components were formed from 302 stainless steel. This material was selected because of its corrosion resistance, good formability, and low cost. During the engine tune-up phase, the combustor cover material was changed to Hastelloy X in order to have adequate life for the tune up test phase.

The manufacturing parts list for the complete engine is contained in Appendix A and photographs of each major engine component are shown in this section.

Table 2 lists the WR33 major subassembly and total engine weight. The engine volume and frontal area were both less than the contract goals.

2.3 PRODUCTION COST ESTIMATE

The other major item accomplished under Task I was the detailed estimation of manufacturing costs (precise within 30 percent) for the WR33 turbojet engine when produced in lots of 1000. This study was conducted in conjunction with the fabrication of the first test engine and was updated near the end of the program in order to include those changes that evolved during the engine tune-up phase.

The results of the manufacturing cost estimate are summarized in Table 3.

The method and rationale used in arriving at the cost estimate and the detailed data are presented in the following paragraphs.

2.3.1 Standard Hours and Direct Cost Dollars

The WRC Manufacturing Engineering Department prepared detailed manufacturing planning on each individual part and subassembly required to produce each engine component. This planning identified each step in the manufacturing process from the preparation of the raw material (shearing of the sheet metal blank, etc.) to the final machining processes and work stations involved.

These process sheets are then forwarded to the Operations Control Department where the costs of raw materials, purchased parts, outside processing, and direct charge dollars are added. This department also estimates the standard hour for set up and the run hours required to complete the task. These raw hour estimates are shown in Table 4.

TABLE 2. DETAILED WEIGHT BREAKDOWN.

	ENGINE COMPONENT	WEIGHT LBS
1.	100 Combustor Assembly	4.0
2.	200 Turbine Shaft Assembly	11.3
3.	300 Turbine Shroud Assembly	0.5
4.	400 Tailcone and Rear Bearing Support Assembly	2.9
5.	450 Rear Bearing Housing Assembly	0.5
6.	600 Combustor Case Assembly	2.8
7.	650 Diffuser and Combustor Cover Assembly	2.8
8.	700 Fuel Transfer and Shaft Seal Housing Assembly	1.7
9.	723 Axial Compressor Rotor (6 Stage)	7.5
10.	725 Axial Compressor Stator (6 Stage)	5.5
11.	750 Compressor Inlet Housing	4.0
12.	850 Nose Cone Assembly	0.2
13.	900 Std Parts	1.3
	TOTAL	45.0

TABLE 3. **MANUFACTURING COST ESTIMATE SUMMARY.**

RFP Goal	**$2000/Engine**
Current Estimate	**$2883/Engine**

Unfactored Cost Element

Cost Element	Current Estimate
Raw Material	$77.00
Purch. Parts	429.00
Outside Processes	212.00
Set-up Hours	259.595
Run Time Hours	56.334

1. Current estimate based on average cost of 5000 units but released in lots of 1000.
2. Costs expressed in CY 1975 economics.
3. To determine set-up hours per engine, divide total set-up by 80.
4. Final engine costs include attrition, inspection, supporting services, tool maintenance, burden, material handling, G and A, and profit.

TABLE 4. WR33 ENGINE SUBASSEMBLY COST SUMMARY.

ITEM DESCRIPTION	Direct Cost Dollars: Raw Mat'l	Direct Cost Dollars: Purch. Parts	Direct Cost Dollars: Outside Process	Standard Labor Hours: Setup Hours	Standard Labor Hours: Run Hours
100 Combustor	$ 9.79	$ 1.2	$54.65	51.250	10.961
200 Turbine Shaft	8.48	289.72	37.00	51.085	10.449
300 Turbine Shroud	23.21	------	21.54	5.150	0.782
400 Turbine Exhaust	7.46	17.48	67.01	40.600	4.688
600 Main Housing	.92	90.06	-----	6.610	1.376
650 Diffuser & Comb Cover Assy	10.35	0.80	29.77	39.350	7.333
700 Fuel Distributor	7.84	66.06	-----	12.050	1.489
719 Axial Comp. Rotor	5.97	-----	10.05	14.550	4.987
726 Axial Comp Stator	6.11	0.08	10.00	17.700	5.820
750 Inlet Housing	5.80	-----	10.05	13.750	3.208
850 Nose Cone	0.63	0.06	4.26	5.650	0.641
950 Engine Assy	-----	28.90	-----	1.850	4.600
TOTALS	86.56	494.36	244.33	259.595	56.334

A detailed summary of all major subassemblies is presented in Tables 5 through 16. Each cost summary table is preceded with a photograph of each major subassembly (Figures 4 through 15).

2.3.2 Estimating and Pricing

The raw hour and direct charge dollar estimates were then forwarded to the contracts department where final engine costs were computed, including attrition, inspection, supporting services, tool maintenance, burden, material handling, G and A, and profit.

Application of the above factors results in an estimated unit price of $2883.00.

The Direct Charge Dollars applicable to the current cost estimate were estimated in CY 1977 economics utilizing recent vendor quotations. They were then converted to CY 1975 economics for comparison purposes utilizing a formula developed from Bureau of Labor Statistics (1977$ ÷ 1.0858 = 1976$, 1976$ ÷ 1.0921 = 1975$). Specifically, the trend developed by SIC Code 3722 (Aircraft Engine and Engine Parts) was utilized.

TABLE 5. PRODUCTION COST ESTIMATE OF BURNER AND TURBINE NOZZLE ASSEMBLY.

PART NAME	Direct Cost Dollars: Raw Mat'l	Direct Cost Dollars: Purch Parts	Direct Cost Dollars: Outside Process	Standard Labor Hours Per Unit: Setup Hours	Standard Labor Hours Per Unit: Run Hours
100 Combustor & Nozzle Assy	$ -	$ -	$10.00	3.800	1.340
100-500 Combustor Braze Assy	-	-	-	3.600	2.475
101 Outer Wall Nozzle Mtg.	2.02	-	8.50	3.500	0.517
102 Mounting Flange	1.37	-	4.25	2.700	0.269
100-503 Vane Assy	-	-	-	0.600	0.945
103 Vane-Press. Side (15)	-	0.240	-	1.700	0.960
104 Vane-Suct. Side (15)	-	0.240	-	1.700	1.11
105 Inner Wall-Nozzle Mtg.	1.01	-	8.50	3.900	0.270
106 Support	1.01	-	8.50	3.400	0.237
107 Shroud	0.31	-	-	1.050	0.137
108 Inner Liner Support	0.45	-	-	3.000	0.360
120 Strip Joint	0.03	-	-	1.000	0.025
100-501 Labyrinth Seal Assy	-	-	-	0.500	0.130
109 Support-Press. Bal. Seal	0.57	-	4.25	3.400	0.228
110 Labyrinth Seal Shaft (2)	-	-	-	0.500	0.050

110-401	Seal Hydro-form	0.76	-	4.26	1.700	0.180
111	Labyrinth Seal-Shaft (2)	-	-	-	0.500	0.050
111-401	Seal Hydro-form	0.56	-	4.26	1.300	0.130
113	Labyrinth Seal-Press. Bal.(2)	0.86	-	2.13	2.100	0.190
114	Rivet (4)	-	0.12	-	-	-
117	Stud (30)	-	0.60	-	-	-
100-502	Liner Assy	-	-	-	1.050	0.167
115	Liner-Inner	0.19	-	-	1.550	0.166
116	Strip	0.03	-	-	1.000	0.025
100-504	Outer Liner Assy	-	-	-	0.500	0.120
118	Outer Wall-Cyl	0.59	-	-	2.200	0.307
119	Strip	0.03	-	-	1.000	0.025
100-505	Assy Start Tube	-	-	-	1.100	0.150
150	Flange	-	-	-	1.600	0.238
151	Tube	-	-	-	1.300	0.160
	TOTALS	9.79	1.2	54.65	51.25	10.961

TABLE 6. PRODUCTION COST ESTIMATE OF TURBINE SHAFT ASSEMBLY.

Part Name	Direct Cost Dollars			Standard Labor Hours/Unit	
	Raw Mat'l	Purch. Parts	Outside Process	Setup Hours	Run Hours
200 - Turbine Shaft Assembly	$ -	$ -	$ -	2.600	1.530
200-500 - Shaft Assy, Braze	-	-	-	7.500	1.466
201 - Turbine Rotor	-	-	-	4.80	2.850
201-900 - Turbine Rotor Casting	-	286.00	-	-	-
203 - Hub, Rear Mach.	-	-	-	6.600	0.689
203-900 - Hub, Rear Casting	1.00	-	3.00	2.000	1.000
204 - Tube, Shaft	0.78	-	2.00	3.450	0.215
200-501 - Fwd. Shaft Assy	-	-	-	0.400	0.150
205 - Hub, Nozzle Mtg.	3.47	-	-	11.485	1.300
206 - Shaft, Fwd.	0.66	-	25.00	6.600	0.724
200-502 - Seal Runner Assy	-	-	-	0.700	0.146
208 - Seal Runner Aft	-	-	-	1.750	0.134
211 - Seal Runner Fwd.	-	-	-	1.750	0.134
220-401 - Runner Hydroform (2)	2.57	-	7.00	1.450	0.111
114 - Rivet (4)	-	0.12	-	-	-
209 - Fuel Nozzle	-	3.60	-	-	-
TOTALS	8.48	289.72	37.00	51.085	10.449

TABLE 7. PRODUCTION COST ESTIMATE OF TURBINE SHROUD ASSEMBLY.

	Part Name	Direct Cost Dollars Raw Mat'l	Purch. Parts	Out-side Process	Standard Labor Hours per Unit Setup Hours	Run Hours
300	Turbine Shroud Assy	$-----	$-----	$15.00	0.100	0.020
301	Flange Mtg.	17.44	-----	6.54	3.750	0.522
302	Shroud Hub	5.77	-----	-----	1.300	0.240
	TOTAL	23.21	-----	21.54	5.150	0.782

TABLE 8. PRODUCTION COST ESTIMATE OF TURBINE EXHAUST HOUSING ASSEMBLY.

Part Name	Direct Cost Dollars			Standard Labor Hours per Unit	
	Raw Mat'l	Purch. Parts	Out-side Process	Setup Hours	Run Hours
400 Tailcone	$-----	$-----	$-------	2.300	0.530
401 Wall, Outer	-----	-----	-------	3.400	0.200
401-401 Wall Hydro.	-----	16.60	-------	-----	-----
400-500 Flange sub-assy	-----	-----	-------	0.400	0.067
404-401 Flange, Heat Shield	-----	-----	-------	2.450	0.141
657-401 Flange, Hyd-roform	0.57	-----	4.25	1.950	0.192
117 Weld Stud (7)	-----	0.14	-------	-----	-----
400-501 Cone Support Subassy	-----	-----	18.00	0.500	0.150
400-502 Support sub-assy	-----	-----	15.00	0.500	0.085
404 Flange, Heat Shield	-----	-----	-------	2.450	0.141
657-401 Flange, Hydr.	0.57	-----	4.25	1.950	0.192
405 Flange, Supp.	1.01	-----	8.50	2.250	0.265
400-503 Cone Subassy	-----	-----	-------	1.100	0.300
402 Vane	0.48	-----	-------	1.900	0.402

403 Wall, Inner	1.58	-----	8.50	4.550	0.330
450 Brg. Hsg. Cup Subassy	-----	-----	-------	1.000	0.184
406 Insulation	-----	0.70	-------	-----	-----
451 Brg. Hsg.	1.92	-----	-------	4.000	0.820
452 Air Shield	0.35	-----	4.25	3.000	0.227
125 Pin, Spring	-----	0.04	-------	-----	-----
453 Heat Shield	0.36	-----	-------	2.000	0.160
500 Strap Assy	-----	-----	-------	0.400	0.043
501 Strap	0.05	-----	-------	1.100	0.044
502 Tail Plug	0.57	-----	4.26	3.400	0.215
TOTAL	7.46	17.48	67.01	40.600	4.688

TABLE 9. **PRODUCTION COST ESTIMATE OF MAIN HOUSING ASSEMBLY.**

Part Name	Direct Cost Dollars			Standard Labor Hours per Unit	
	Raw Mat'l	Purch. Parts	Out-side Process	Setup Hours	Run Hours
600 Case, Combust. Assy	$-----	$-----	$-------	0.400	0.133
606 Case, Combust.	-----	-----	-------	2.160	0.203
606-401 Case, Hydroform	-----	89.25	-------	-----	-----
610 Ring Assy	-----	-----	-------	0.400	0.270
608 Ring	0.90	-----	-------	2.950	0.512
117 Weld Stud (3)	-----	0.72	-------	-----	-----
609 Bracket (3)	0.02	-----	-------	0.500	0.162
607 Rivet	-----	0.09	-------	0.200	0.096
TOTAL	0.92	90.06	-------	6.610	1.376

TABLE 10. PRODUCTION COST ESTIMATE OF DIFFUSER AND COMBUSTOR COVER

		Direct Cost Dollars			Standard Labor Hours per Unit	
Part	Name	Raw Mat'l	Purch. Parts	Outside Process	Setup Hours	Run Hours
650	Diffuser & Comb. Cover Assy	$ -	$ -	$ -	1.700	0.815
650-500	Diffuser Assy	-	-	-	2.000	0.788
651	Cover Diffuser	3.91	-	4.25	5.750	0.897
652	Support (6)	0.06	-	-	0.800	0.192
653	Outer Wall Diffuser	0.46	-	-	1.700	0.195
655	Inner Wall Diffuser	1.37	-	4.25	5.750	1.720
117	Weld Stud (22)	-	0.44	-	-	-
650-501	Burner Cover Assy	-	-	-	0.500	0.297
658	Cover, Burner	1.57	-	8.50	4.000	0.497
650-502	Labyrinth Seal Assy	-	-	-	0.500	0.297
657	Support	-	-	-	2.700	0.200
657-401	Support Hydroform	0.57	-	4.25	1.950	0.192
659	Labyrinth Seal	-	-	-	0.500	0.017
110-401	Labyrinth Hydroform	0.38	-	4.26	1.700	0.090
660	Labyrinth Seal	-	-	-	0.500	0.017
1	Labyrinth Hydroform	0.38	-	4.26	1.700	0.090
114	Rivet	-	0.36	-	-	-
661	Ignition Boss	0.60	-	-	5.250	0.360
662	Brkt. - Engine Mtg. (3)	1.05	-	-	2.350	0.669
	TOTAL	10.35	0.80	29.77	39.35	7.333

TABLE 11. MANUFACTURING COST ESTIMATE OF FUEL DISTRIBUTION HOUSING ASSEMBLY.

		Direct Cost Dollars			Standard Labor Hours per Unit	
Part	Name	Raw Mat'l	Purch. Parts	Outside Process	Setup Hours	Run Hours
700	Hsg-Fuel Dist. Assy	$ -	$ -	$ -	0.700	0.250
715	Fitting Air	0.17	-	-	2.500	0.196
704	Fuel Housing	7.67	-	-	7.500	0.963
133	Carbon Seal	-	12.10	-	-	-
134	O-Ring	-	0.18	-	-	-
135	O-Ring	-	1.24	-	-	-
705	Brg. Thrust	-	28.50	-	-	-
707	Locknut	-	-	-	1.350	0.080
707-401	Locknut	-	1.49	-	-	-
411	Brg - Rear Housing	-	17.80	-	-	-
708	O-Ring	-	0.10	-	-	-
709	Runner, Seal Fuel	-	4.65	-	-	-
	TOTAL	7.84	66.06	-	12.050	1.489

TABLE 12. PRODUCTION COST ESTIMATE OF AXIAL COMPRESSOR ROTOR.

		Direct Cost Dollars			Standard Labor Hours per Unit	
	Part Name	Raw Mat'l	Purch. Parts	Outside Process	Setup Hours	Run Hours
719	Axial Rotor Mach.	$ -	$ -	$ -	8.100	1.720
719-900	Rotor Casting	5.80	-	10.05	2.000	3.000
714	Spacer	0.17	-	-	4.450	0.267
	TOTAL	5.97	-	10.05	14.550	4.987

TABLE 13. PRODUCTION COST ESTIMATE OF AXIAL COMPRESSOR STATOR.

	Part Name	Direct Cost Dollars Raw Mat'l	Purch. Parts	Outside Process	Standard Labor Hours Per Unit Setup Hours	Run Hours
726	Axial Comp. Stator	$ -	$ -	$ -	15.300	2.120
726-900	Stator Casting	5.80	-	10.00	2.000	3.000
778	Shim (2)	0.31	-	-	0.400	0.700
125	Dowell (4)	-	0.08	-	-	-
	TOTALS	6.11	0.08	10.00	17.700	5.820

TABLE 14. PRODUCTION COST ESTIMATE OF INLET HOUSING.

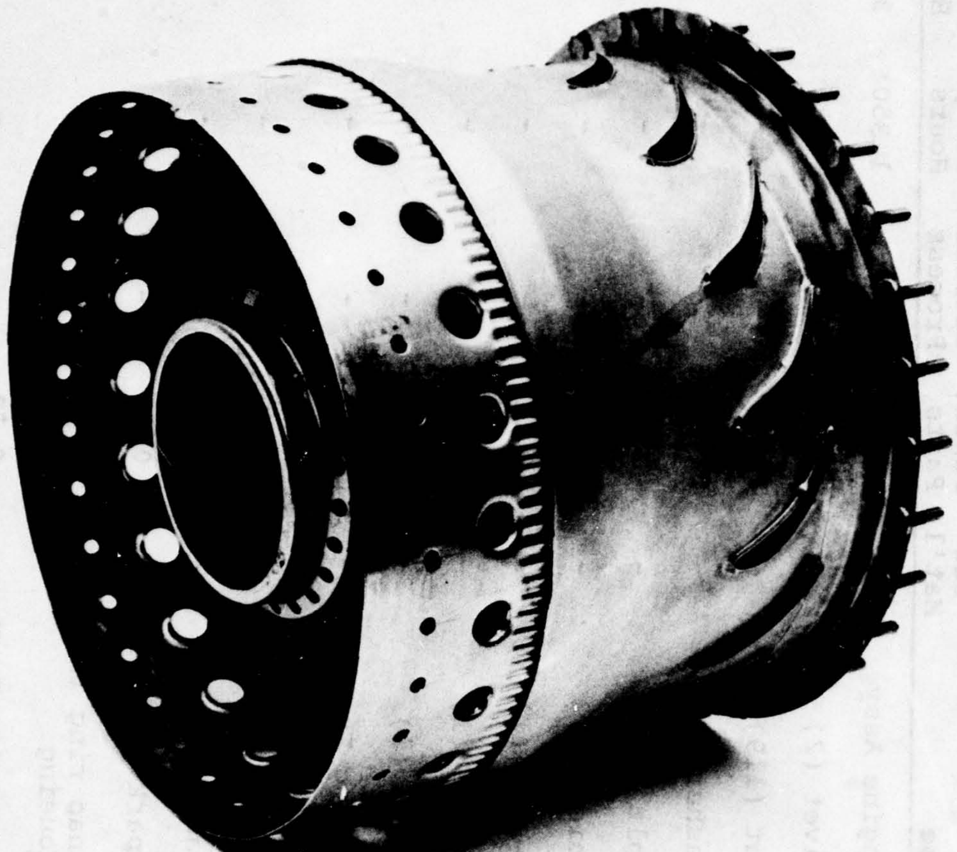
	Part Name	Direct Cost Dollars: Raw Mat'l	Direct Cost Dollars: Purch. Parts	Direct Cost Dollars: Outside Process	Standard Labor Hours Per Unit: Setup Hours	Standard Labor Hours Per Unit: Run Hours
750	Comp. Inlet-Mach.	$ -	$ -	$ -	11.750	1.208
750-900	Casting-Inlet Hsg.	5.80	-	10.05	3.000	2.000
	TOTAL	5.80	-	10.05	13.750	3.208

TABLE 15. PRODUCTION COST ESTIMATE OF INLET NOSE CONE ASSEMBLY.

Part Name		Direct Cost Dollars: Raw Mat'l	Direct Cost Dollars: Purch. Parts	Direct Cost Dollars: Outside Process	Standard Labor Hours per Unit: Setup Hours	Standard Labor Hours per Unit: Run Hours
850	Inlet Nose Cone Assy	$ -	$ -	$ -	0.500	0.098
851	Cone	0.57	-	4.26	4.150	0.252
852-500	Strap Assy	-	-	-	0.400	0.090
852	Strap (3)	0.06	-	-	0.600	0.201
853	Pin (3)	-	0.06	-	-	-
	TOTAL	0.63	0.06	4.26	5.650	0.641

TABLE 16. **PRODUCTION COST ESTIMATE OF MAIN ENGINE ASSEMBLY.**

Part	Name	Direct Cost Dollars			Standard Labor Hours per Unit	
		Raw Mat'l	Purch. Parts	Outside Process	Setup Hours	Run Hours
950	Engine Assy	$ -	$ -	$ -	1.350	3.600
114	Rivet (2)	-	0.06	-	-	-
122	Nut (119)	-	22.61	-	-	-
126	Washer (75)	-	1.50	-	-	-
127	Bolt (24)	-	0.48	-	-	-
129	Union (1)	-	1.38	-	-	-
130	O-Ring (1)	-	0.11	-	-	-
131	O-Ring (1)	-	0.10	-	-	-
132	Union (1)	-	1.49	-	-	-
	O-Ring Fuel Hsg. (3)	-	0.36	-	-	-
137	Screw (7)	-	0.35	-	-	-
138	Washer (3)	-	0.12	-	-	-
149	Spark Plug (1)	-	0.13	-	-	-
91-407	Snap ring Inlet Housing	-	0.08	-	-	-
412	Snap ring R. Hsg	-	0.13	-	-	-
Engine Test & Acceptance		-	-	-	0.500	1.000
	TOTALS	-	28.9	-	1.850	4.600



C-4762
A-0030

Figure 4. Burner and Nozzle Assembly - P/N 100

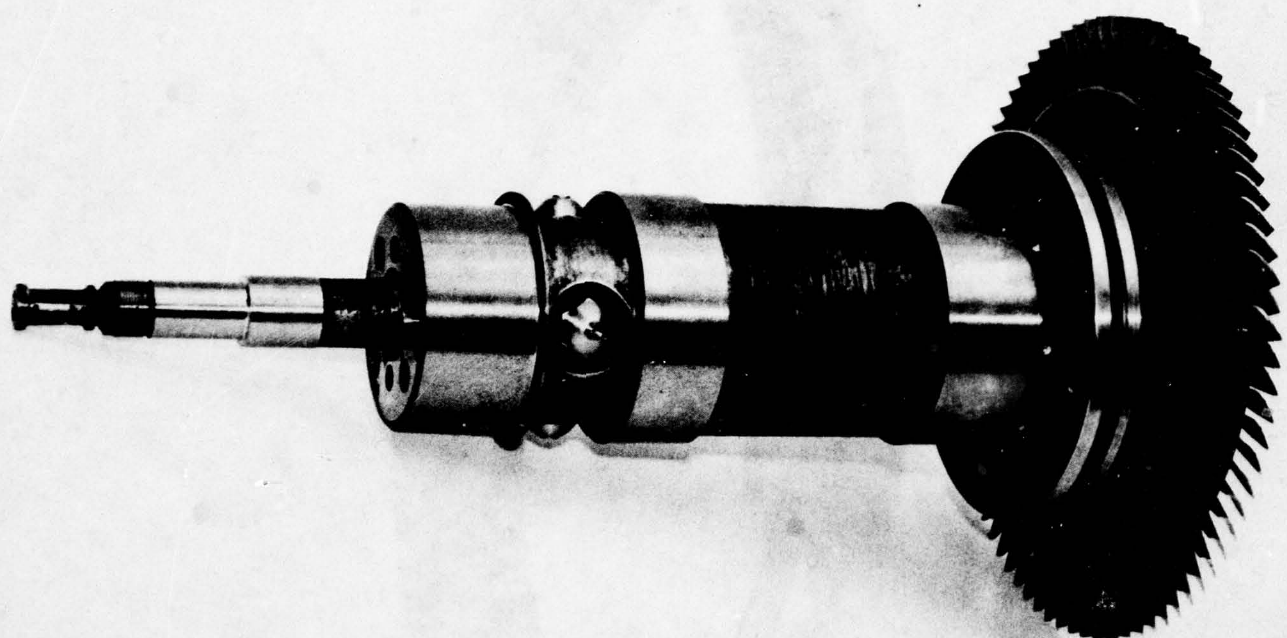


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Figure 5. Turbine Shaft Assembly - P/N 200

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A–0028

Figure 6. Turbine Shroud Assembly - P/N 300

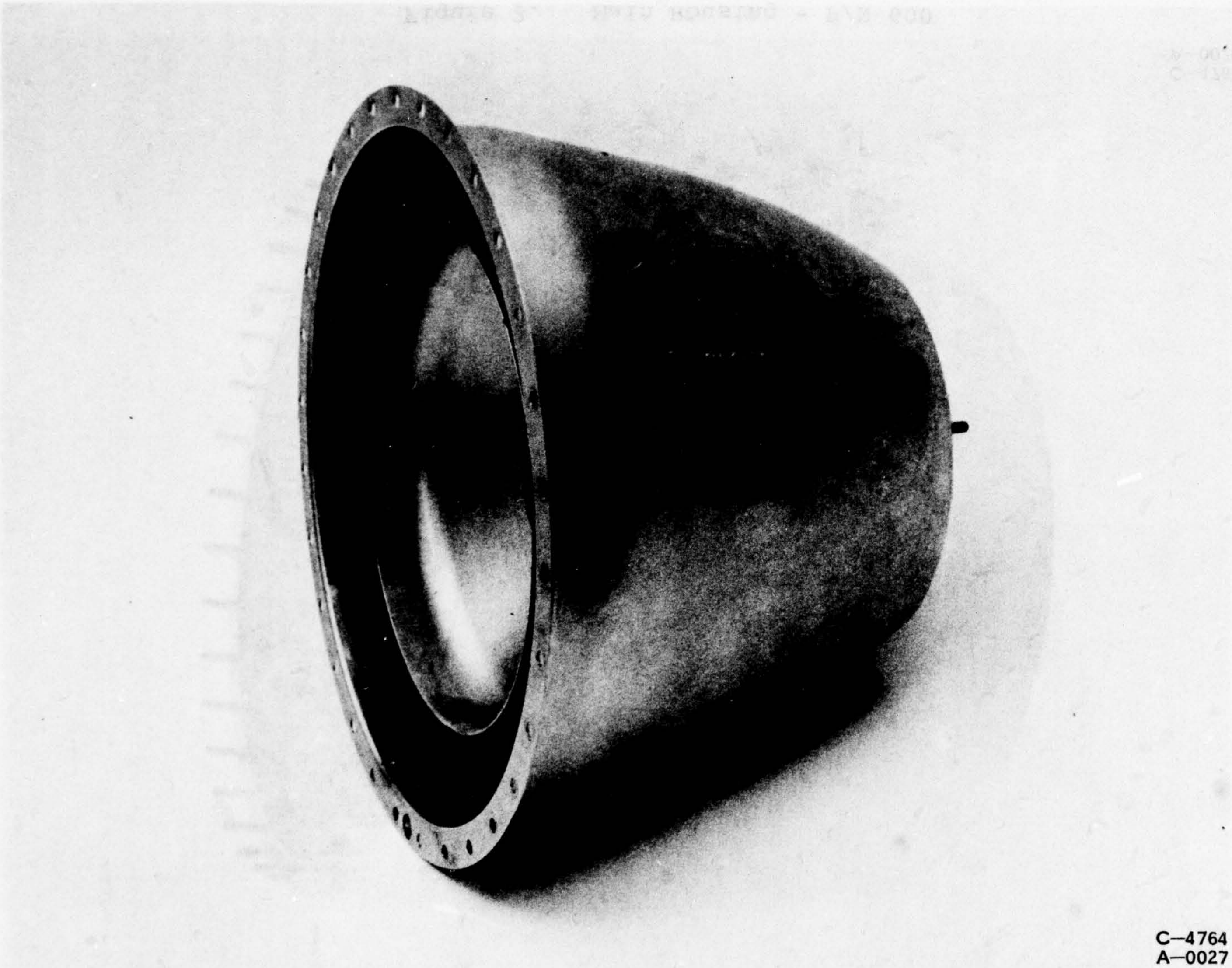

Figure 7. Turbine Exhaust Housing Assembly - P/N 400

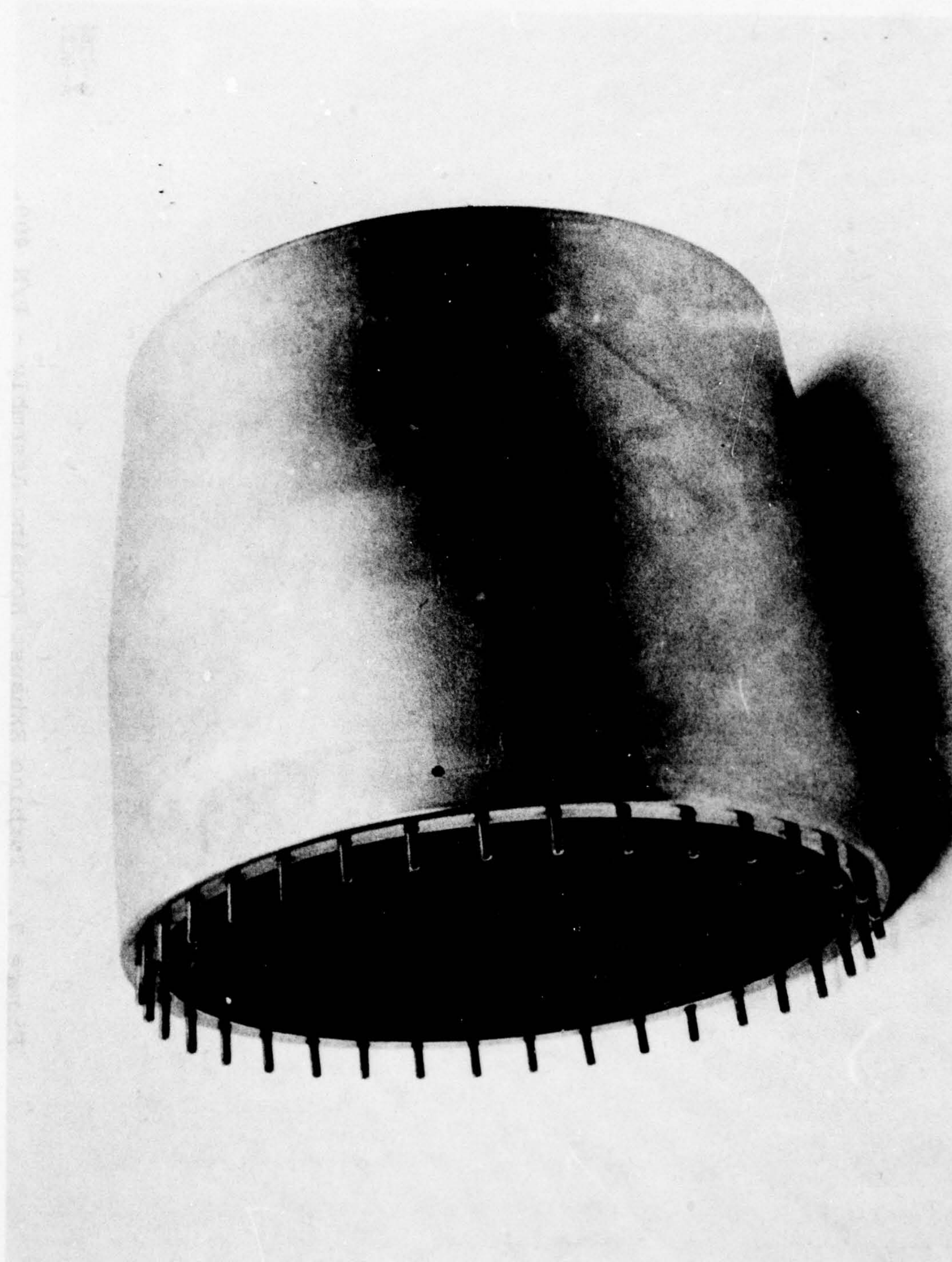

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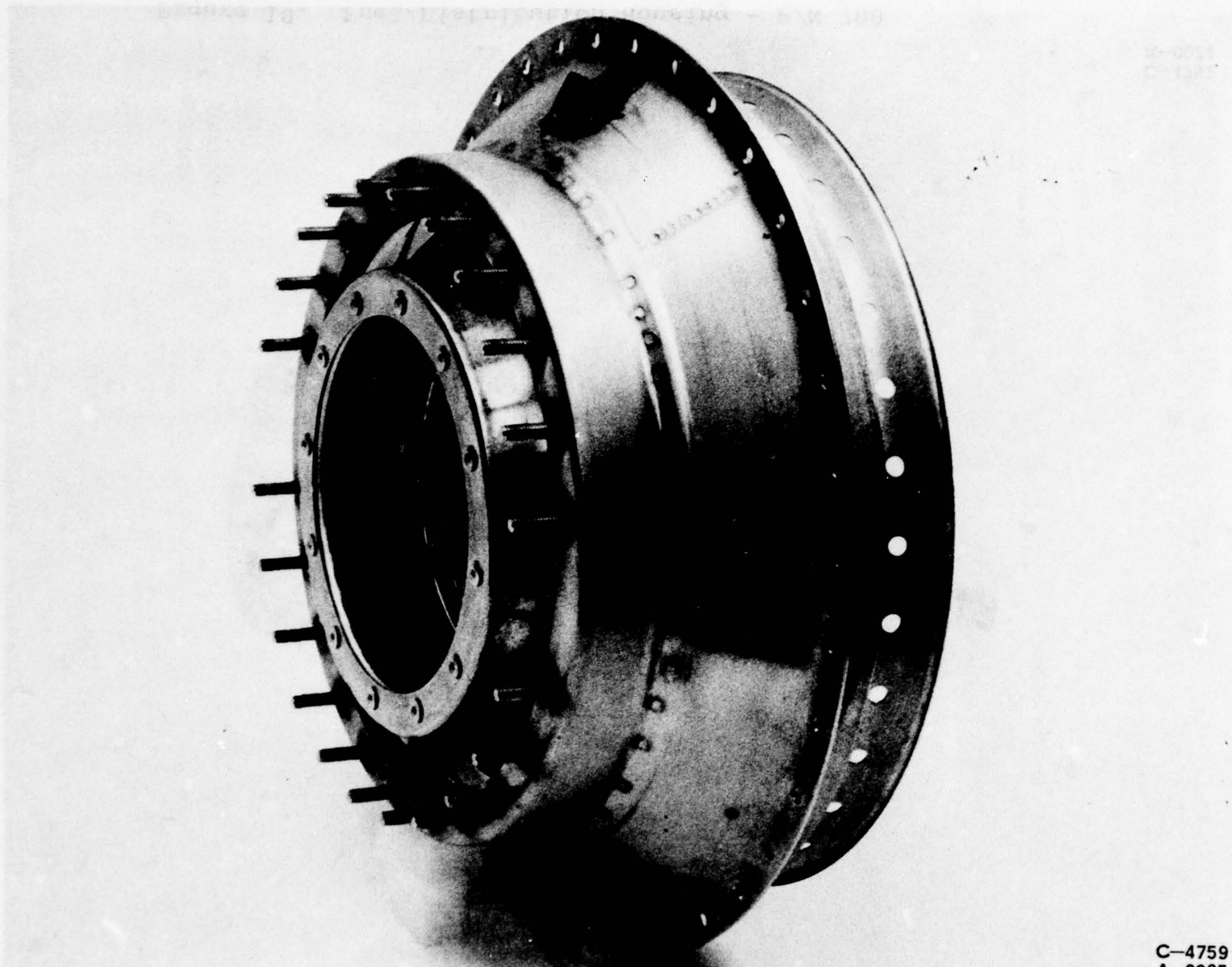
Figure 8. Main Housing - P/N 600



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A—0025

Figure 9. Diffuser and Combustor Cover - P/N 650

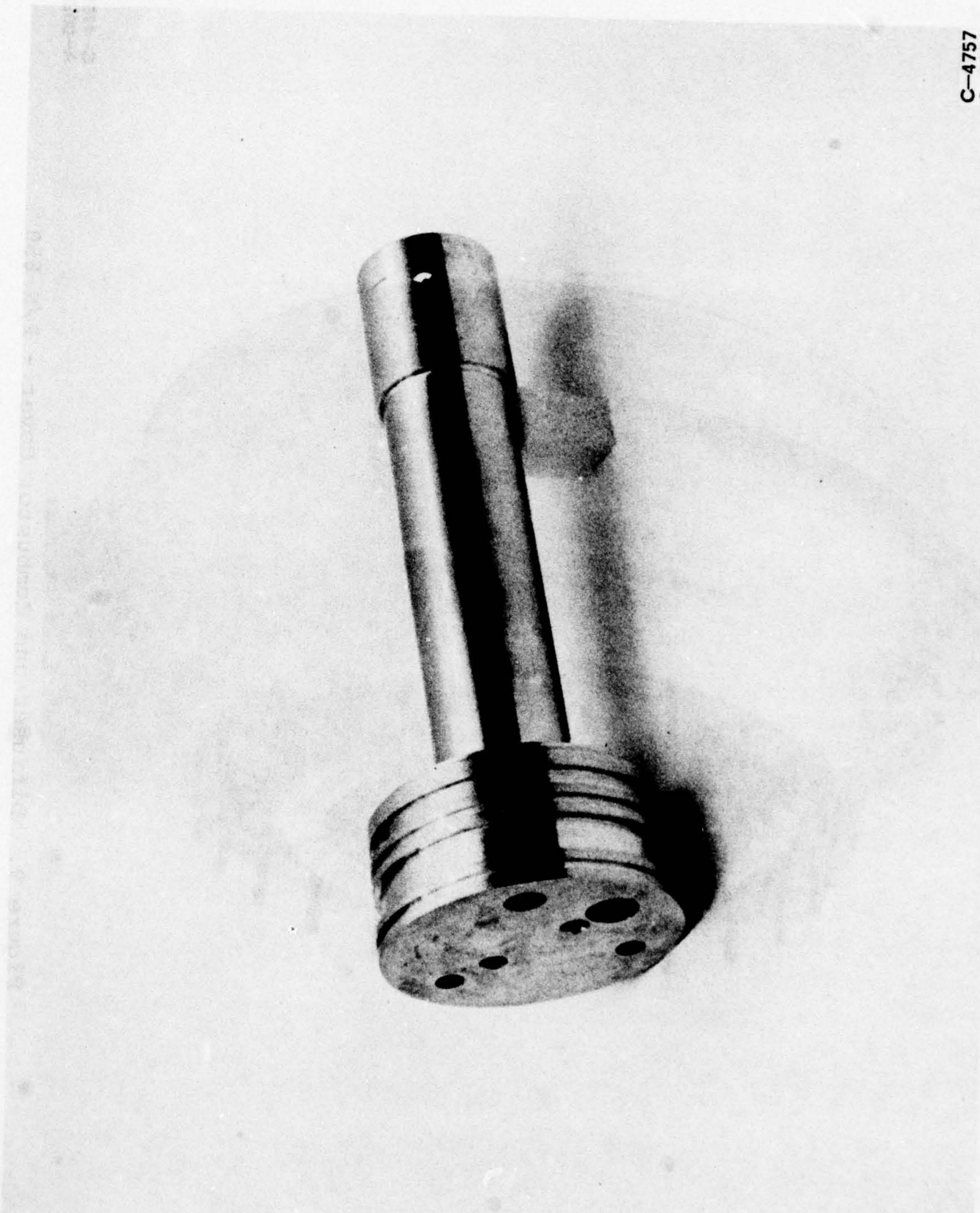

Figure 10. Fuel Distribution Housing - P/N 700

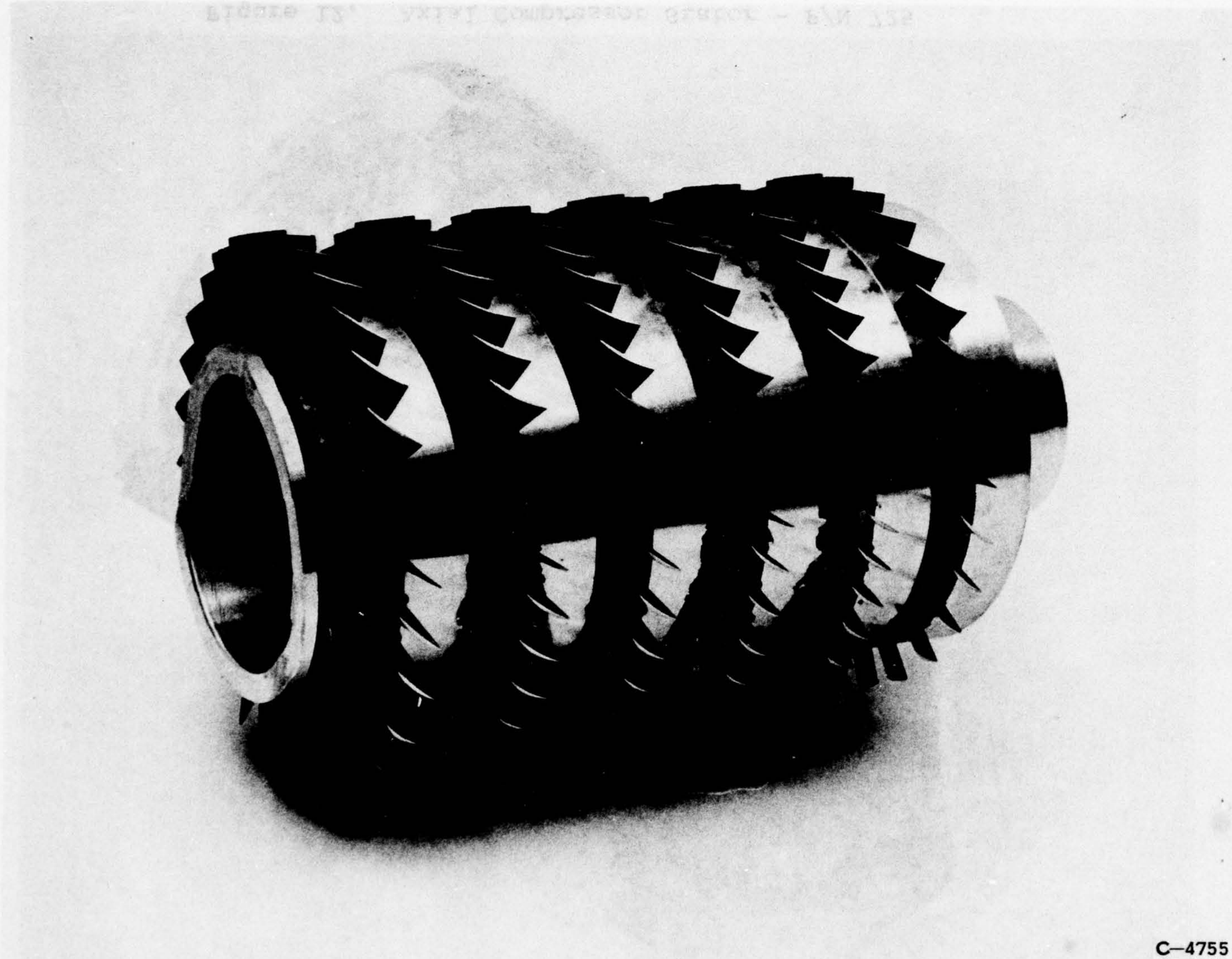

Figure 11. Axial Compressor Rotor - P/N 723

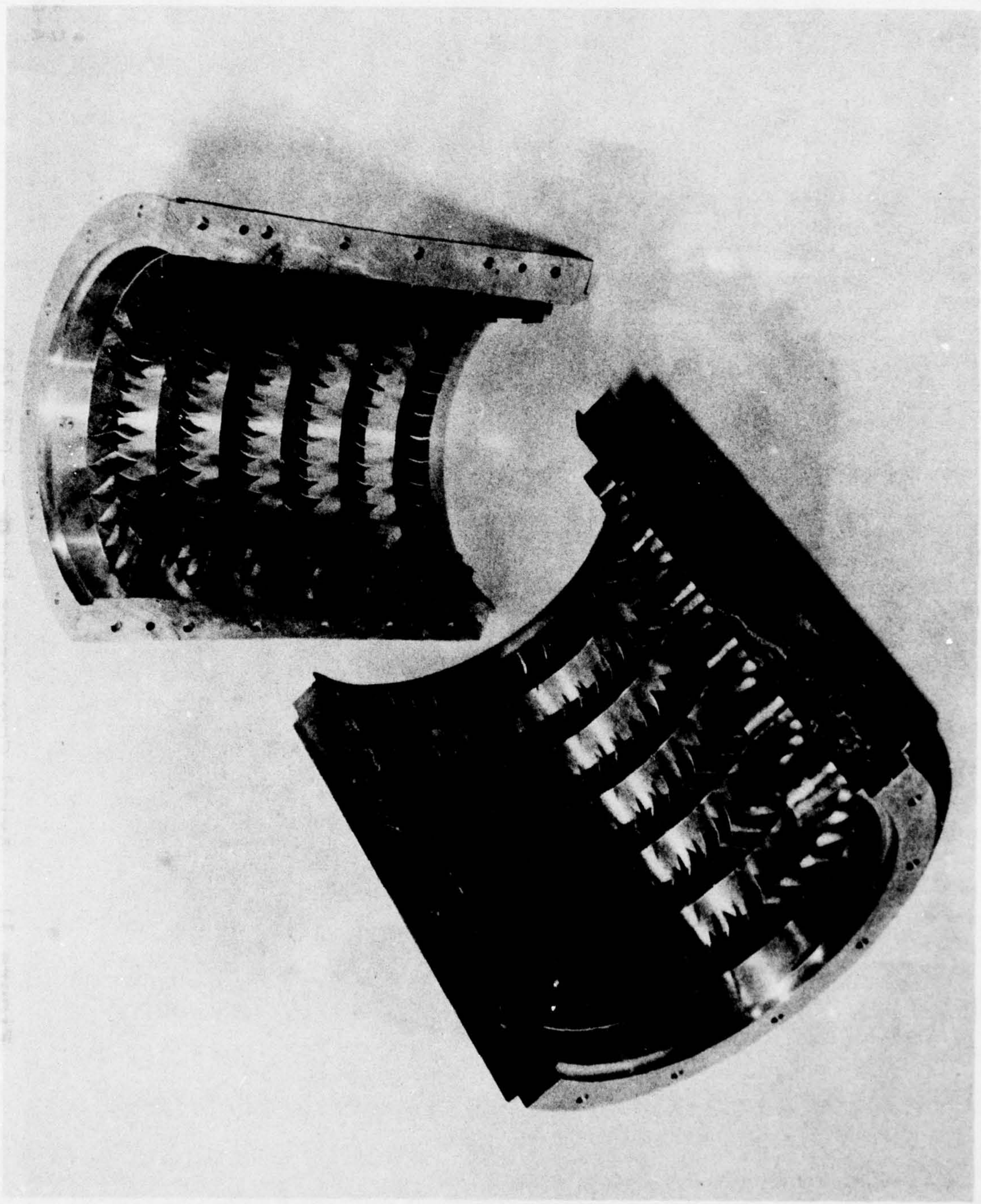

Figure 12. Axial Compressor Stator - P/N 725

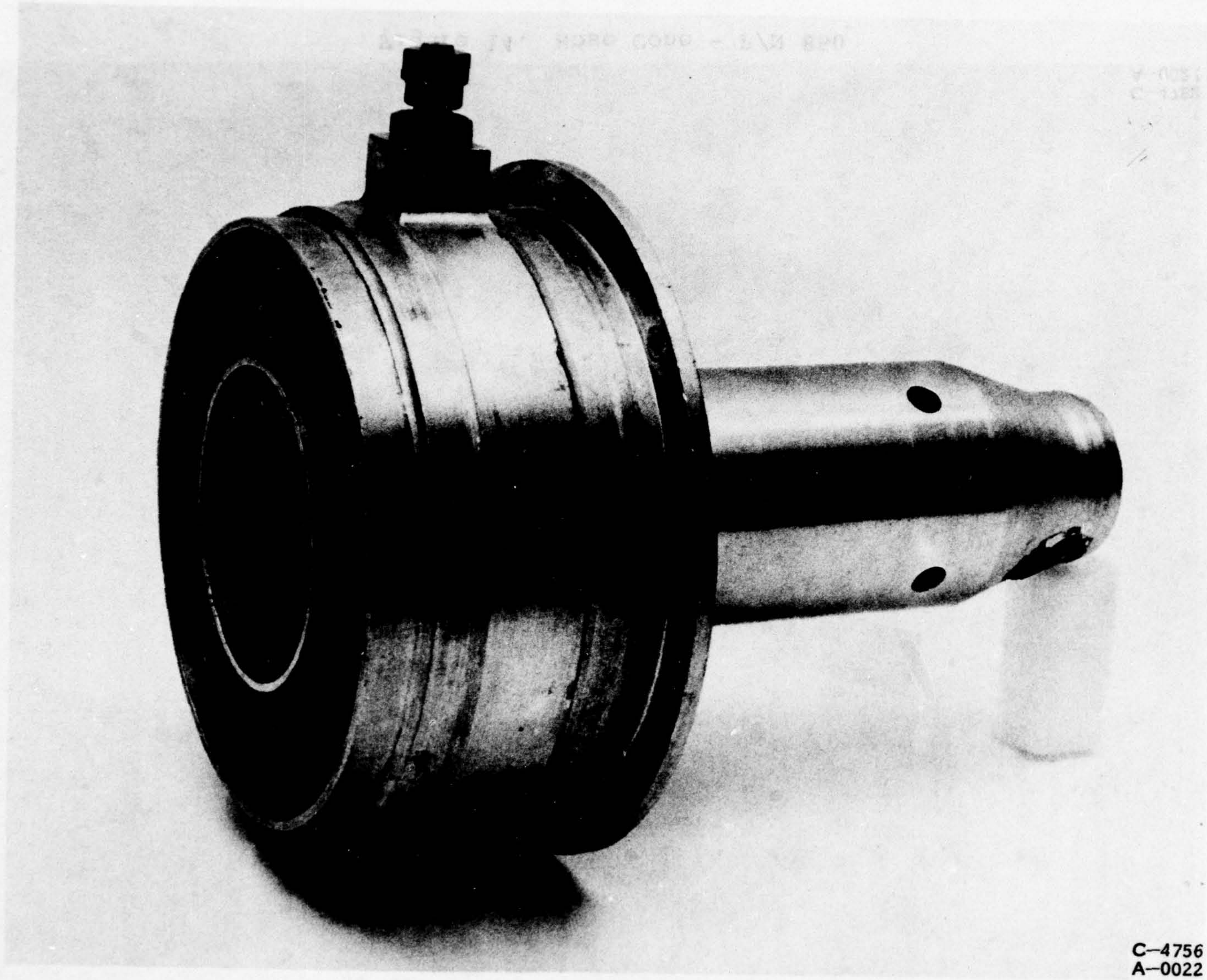

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Figure 13. Inlet Housing - P/N 750

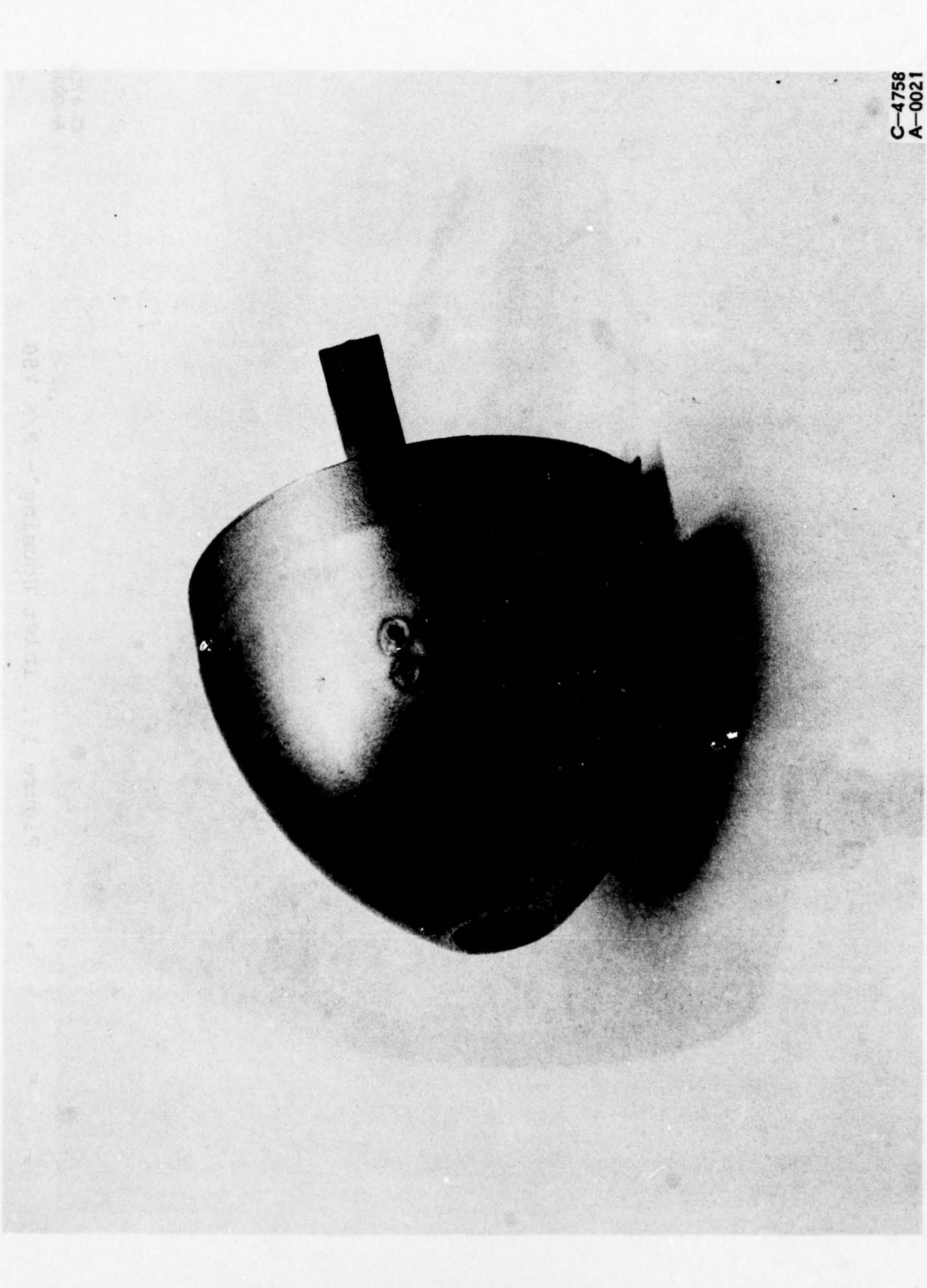


Figure 14. Nose Cone - P/N 850

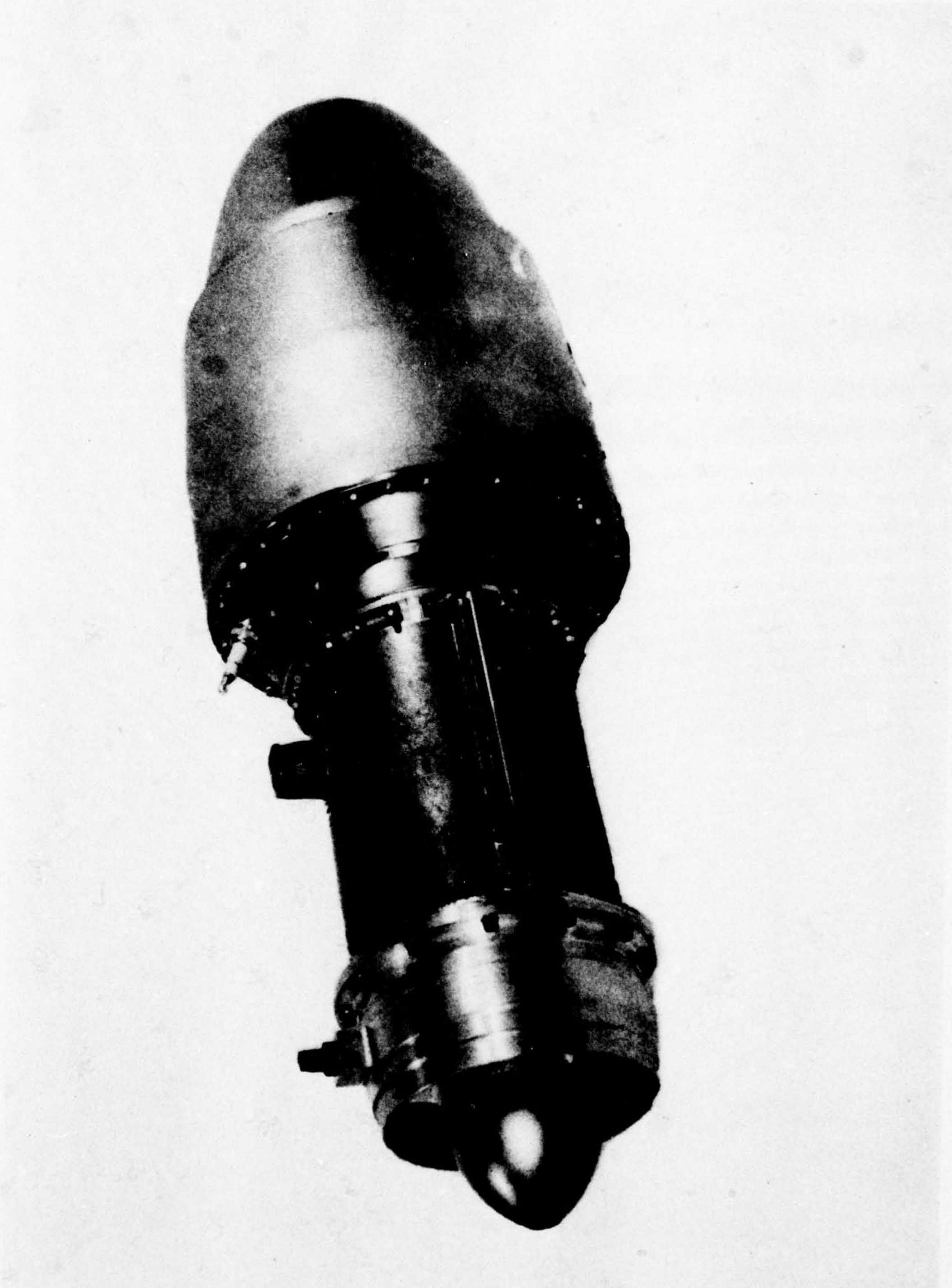


Figure 15. Engine Assembly - P/N 950

SECTION 3

TASK II COMPONENT AND TUNING TESTS

Task II of the Low Cost Expendable Engine program involved component level testing of the axial compressor and engine tuning testing on the first complete WR33 engine. This section of the report presents the results of this effort.

3.1 COMPRESSOR TESTING

Several compressor rig tests were conducted under this program. The initial test was conducted on a five stage integral axial compressor designed for a somewhat larger engine and was intended only to provide design verification data for the WR33 design. This compressor was operated up to design speed and for a test duration just under five hours before suffering a blade fatigue failure.

Although this life was essentially ten times the engine life goal, it was felt that the test hardware used for the rig testing and engine tuning testing of the WR33 would have to have a longer life in order to complete the amount of testing that was anticipated. Because of this, a bicast compressor construction was used for the WR33 test hardware which required about a 50 percent higher level of casting physical properties. The initial rotor castings did not achieve the required strength and ductibility levels and two compressor failures occured during the course of the rig testing.

The first axial compressor rotor failure occurred during the initial compressor rig test in January 1977. The rotor drum failed at about 93 percent of rated speed during the rig check-out. Failure occurred by rupture though two drilled passages which were intended to conduct fourth-stage air to the rear bearing for cooling. The energy from the rotor burst also destroyed the stator casting.

It was concluded that the rotor failed below design speed because the material properties of the casting were insufficient to carry the loads produced by the design features of the inserted blades and the location and configuration of the cooling air holes.

Two solutions to the problem were proposed. One was to continue working on foundry techniques to improve the casting

physical properties. The second was to revise the design configuration to eliminate the air cooling passages and the mounting bolt holes in the rear hub section so that the required casting strength levels could be reduced. The rotor bore ID was also reduced to eliminate protrusion of the blade stock through the drum wall after final machining.

As an interium step, one integral compressor rotor casting was hot isostatic pressed at a temperature of 910-930 ^{o}F and 15,000 psi for 3 hours. This rotor was proof spun and failed at 104 percent of design speed. The HIPing had improved the strength levels somewhat but the level of ductility was still unacceptable.

A proof spin speed of 114 percent of design speed was established as the requirement that was to be met before proceeding with compressor rig testing. This speed represents 130 percent of design centrifugal stress and it was felt that this would provide adequate margin for the rig testing and engine demonstration program.

Continued refinement of the casting process resulted in rotors which met the proof spin requirement and no further drum failures occurred during the program.

During the compressor rig testing in July of 1977, a blade separated from the rotor which in turn sheared all of the rotor blading. A detailed examination of the rotor showed that many of the blade roots had melted during the casting process and that these blades were retained by only a fraction of the cross-sectional area of the blade. Higher than anticipated stresses due to the reduced cross-section coupled with the forces associated with running in surge was the cause of failure. At the time of failure, the rotor had accumulated 7.9 hours of operation and was running at 97 percent of design speed.

A suitable method of coating the blade root to insulate it such that it did not melt during the casting process was evolved. Rotors using this technique were successfully tested on the compressor test rig to 105 percent of design speed and were used for both the engine tune-up testing and demonstration testing without failure.

3.1.1 Compressor Test Rig

An existing compressor rig located in test cell A-1 was renovated and regeared to accommodate the WR33, six-stage axial compressor rotor.

The general arrangement of the compressor rig test facility is shown in Figure 16. Outside air enters the inlet cupola and passes through the ASME airflow measuring nozzle. Inlet total pressure and temperature and static pressure at the nozzle throat are measured. The accuracy of airflow measurement using this system is ±1/2 percent. The flow then passes through an insulated 8 inch diameter cylindrical duct and a 90° bend to a serrated orifice plate. This plate is designed to reduce drive horsepower requirements and provide an additional mass flow rate check for high flow rate compressors, by running choked while providing an inlet pressure drop. The orifice plate also serves as a thorough flow mixer; hence, a more uniform compressor inlet flow with a thinner boundary layer results.

On the downstream side of the orifice plate, the flow is directed through a 5 inch diameter pipe to the compressor inlet measuring station. This station is approximately 10 inches upstream from the inlet duct-compressor interface. Total pressure and temperature and static pressure measurements are made at this location. This is the point to which all compressor performance is referenced. The air then passes through the compressor into a large discharge plenum where temperature and static pressure are measured. The air then leaves the discharge plenum through two opposing headers which merge into one larger duct which guides the air through the discharge throttling valves. There are two throttling valves, a main valve, and a vernier valve. The vernier valve permits the fine adjustments necessary when operating near the compressor surge line. From this point the flow discharges to the atmosphere.

The drive system for the compressor rig consists of two 413 cu. in. Chrysler industrial gasoline engines. The engines may be operated either separately or together. When operated together, throttles are adjusted to produce equal intake manifold pressures in both engines. When only one engine is being run, the other is declutched and out of the system. Each drive engine is coupled to a drive shaft via cog belts. The cog belt pulleys step up the engine speed by a ratio of 1.6:1. The drive shaft is then coupled to the input shaft of the gearbox through a shear coupling.

Except for special instrumentation, all probes and sensors are plugged into a boom in the cell. The boom leads between the test cell and the console outside the cell are permanently installed. Thermocouple leads terminate at a Lewis thermo-

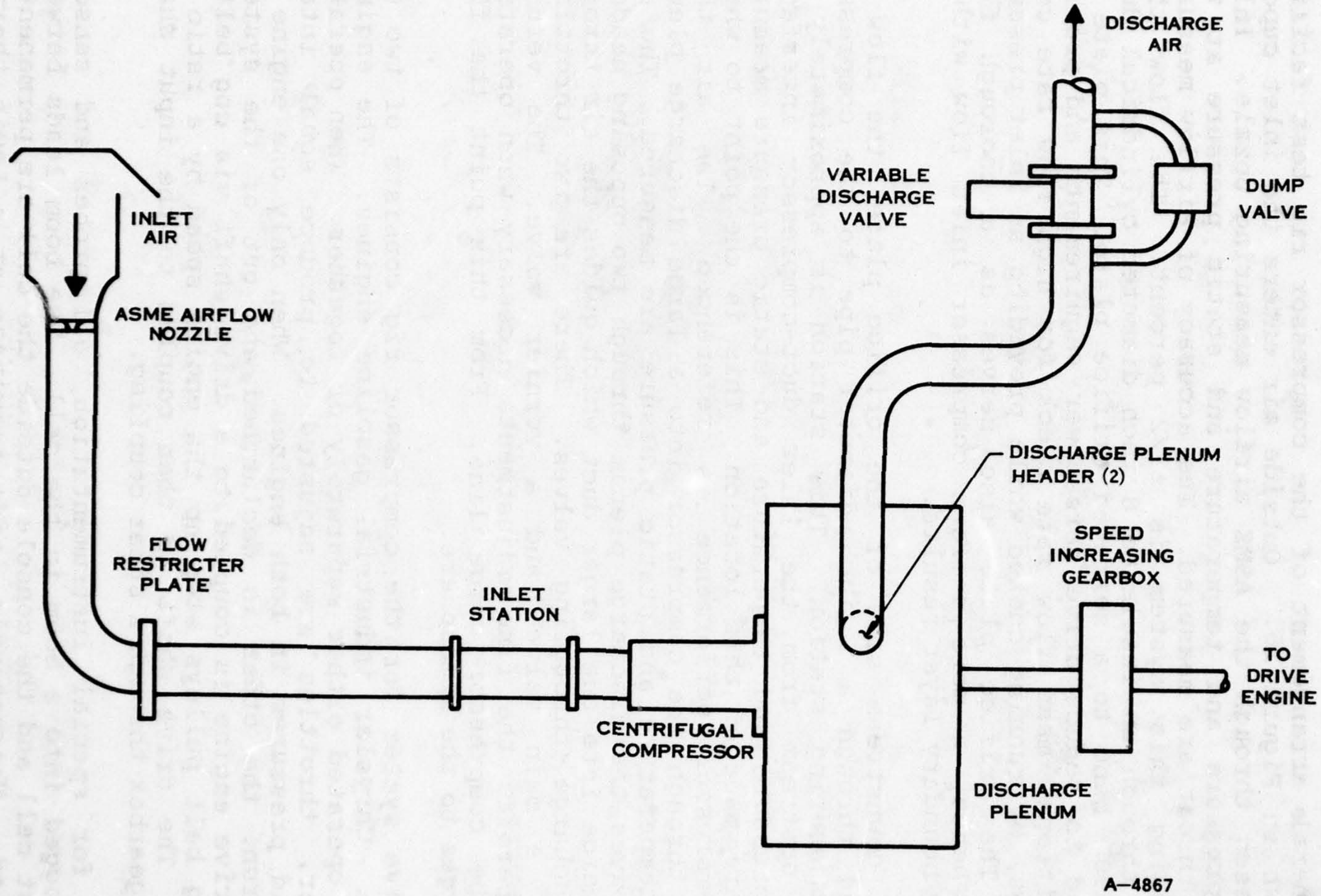


Figure 16. Compressor Test Rig Facility

couple switch which is connected to a digital readout device capable of 1^{o}F accuracy. Pressures from the cell terminate at a panel behind the console and are attached either through manifolds or directly to manometer columns, pressure gauges, recorders, etc.

3.1.2 Compressor Rig Instrumentation

The compressor rig test hardware and instrumentation location are shown in Figure 17. The test rig was designed to use the standard low cost engine compressor, stator and diffuser hardware. The stator housing was machined to incorporate the interstage instrumentation as shown.

Static pressure at the ASME nozzle throat was measured on an inclined water manometer. Pressures at the compressor inlet was measured on a vertical manometer using a fluid with specific gravity 2.95 times that of water. All other pressures were measured in inches of mercury on either conventional vertical manometers or, in the case of the higher pressures, on a Wallace & Tiernan Bourdon Tube Dial Gauge. All static pressure taps are sharp edged 0.020 inch diameter holes drilled perpendicular to the local stream-wise direction. Total pressure probes were of the single tube impact variety and constructed with sharp, chamfered, burr-free inlets, except the interstage probes which were keil head probes.

All thermocouples were of the copper-constantan, bare wire type, with the exception of those in the discharge plenum headers which were half shielded. The compressor rig gearbox and drive system is shown in Figure 18 and the control console is shown in Figure 19.

3.1.3 Compressor Rig Test Procedure

The compressor performance maps were generated by recording test data along constant speed lines. Each speed line consisted of a minimum of four data points; wide open throttle, surge, and two intermediate airflows. The speed lines recorded were 40, 50, 60, 70, 80, 90, 95, 100 and 105 percent corrected speed.

3.1.4 Compressor Rig Test Results

The large scale compressor tests, although not a part of this program, were extremely important in that they were the

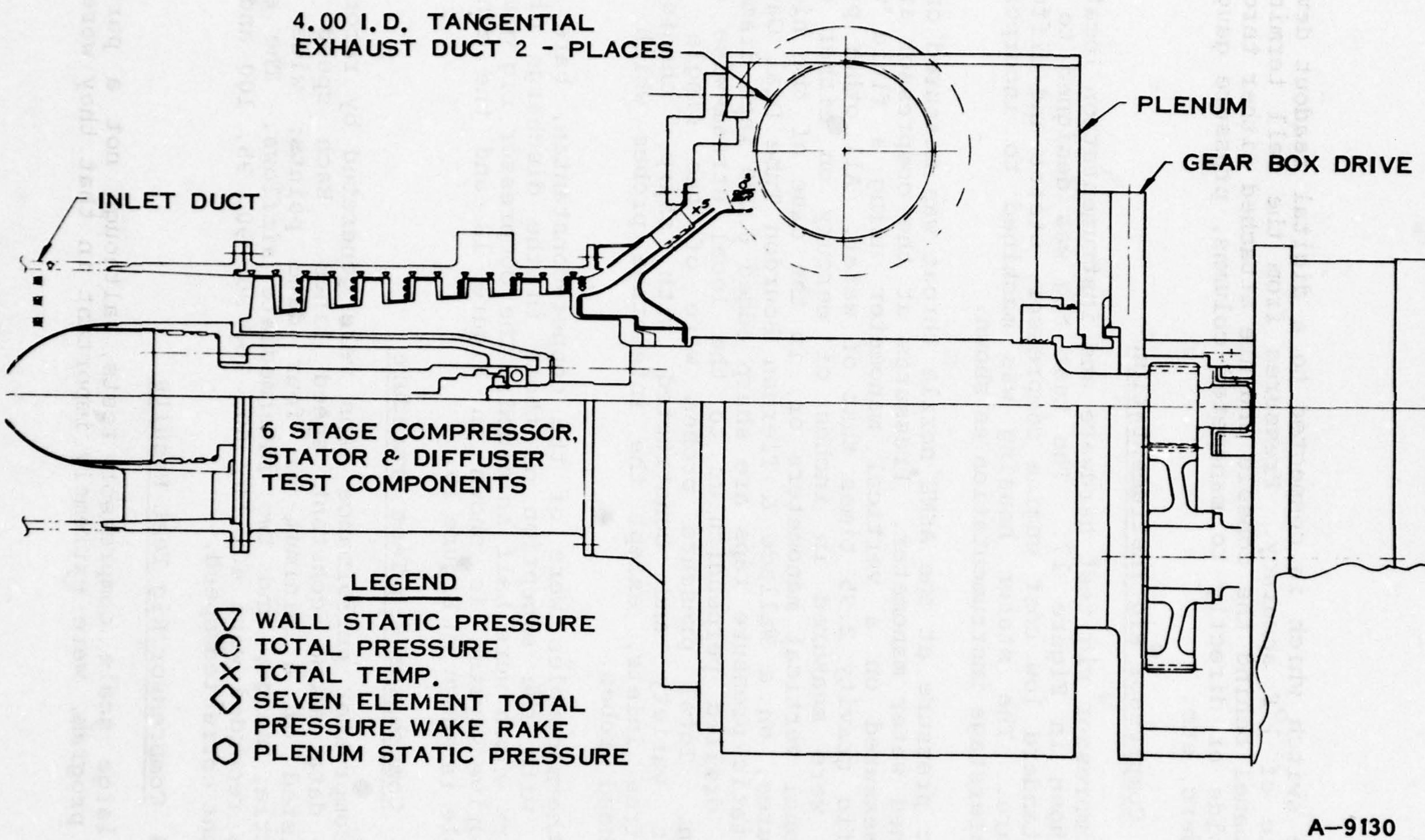


Figure 17. Axial Compressor Test Fixture

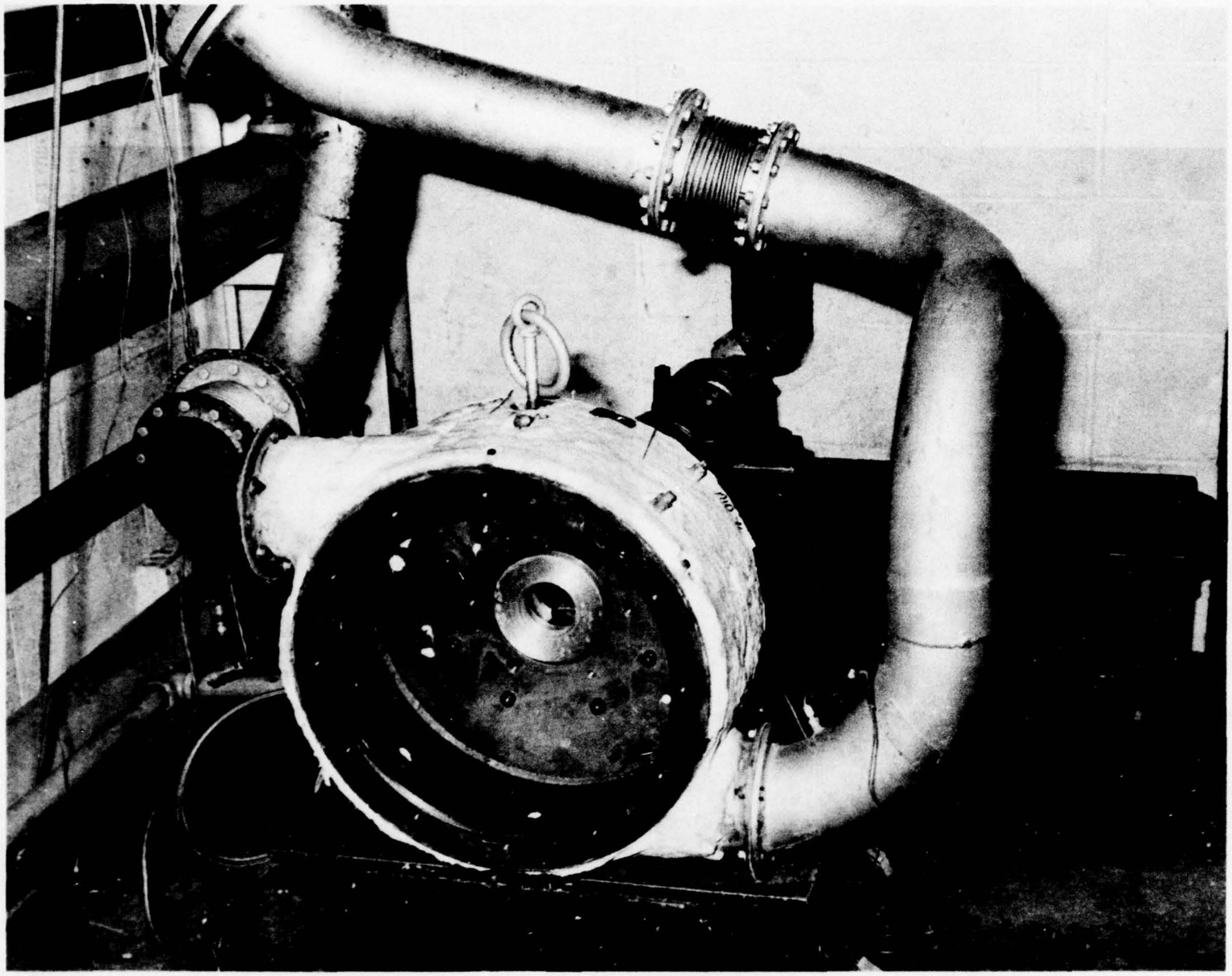


Figure 18. Compressor Test Rig Plenum, Gearbox and Drive Shaft Assembly

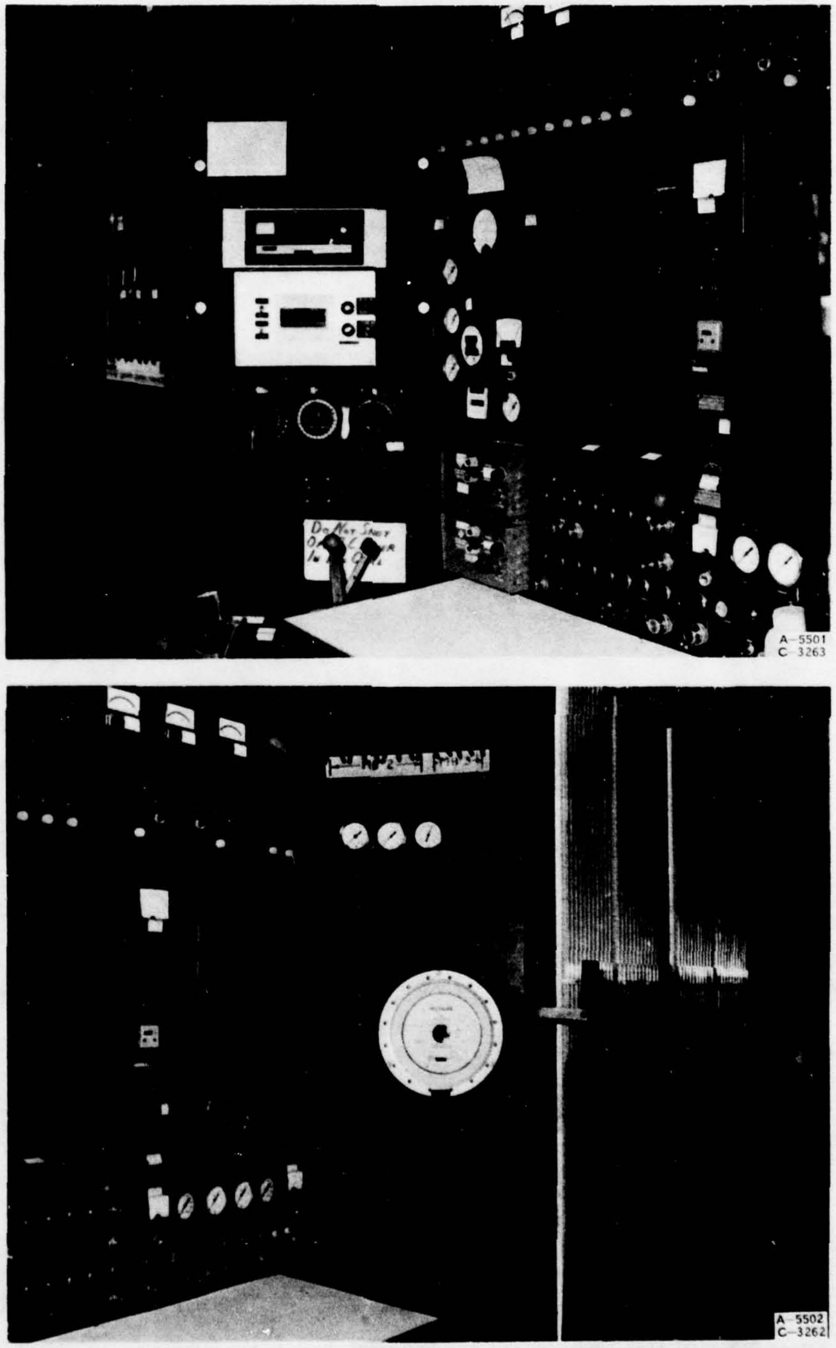


Figure 19. Compressor Test Rig Plenum Control and Instrumentation Console

first substantiation of the low cost integral multi-stage axial compressor concept.

During the period of testing, data was taken up to and including 100 percent design mechanical speed. Due to the high ambient air inlet temperature during the time of the testing, this resulted in corrected speed data up to 95 percent speed. Photographs of the test compressor after test are shown on Figures 20 and 21.

The compressor performance that was obtained is shown on Figure 22. The pressure ratio and efficiency shown on this map are based on mid-stream instrumentation. These probes were connected to an automatic data acquisition system and were recorded at each data point. The remainder of the instrumentation was connected to manometer banks and was recorded manually at selected data points only. The two solid points represent the actual performance when all of the data is considered which includes the effects of wakes and the wall boundaries.

The performance shown on this map exceeded our expectations considering the quality of the test hardware and, based on this performance, WRC felt confident to proceed with the tooling for the actual WR33 prototype engine compressor.

Figure 23 shows the actual WR33 engine compressor installed on the compressor test rig. This compressor was sucessfully tested to 105 percent of design speed and the performance maps are shown on Figures 24 and 25 for total-to-total and total-to-static conditions. The measured performance on the rig fell slightly below our predictions, however, when tested in the engine, the compressor performance was very near our expectations. Because of the nature of the prototype tooling used in the manufacture of the bicast axial compressor rotor and stator there was a variation in blade twist and angle between the various rotors.

3.2 ENGINE TUNING TEST

The second phase of Task II of the low cost expendable engine program involved a tuning and engine adjustment period using a complete WR33 turbojet engine.

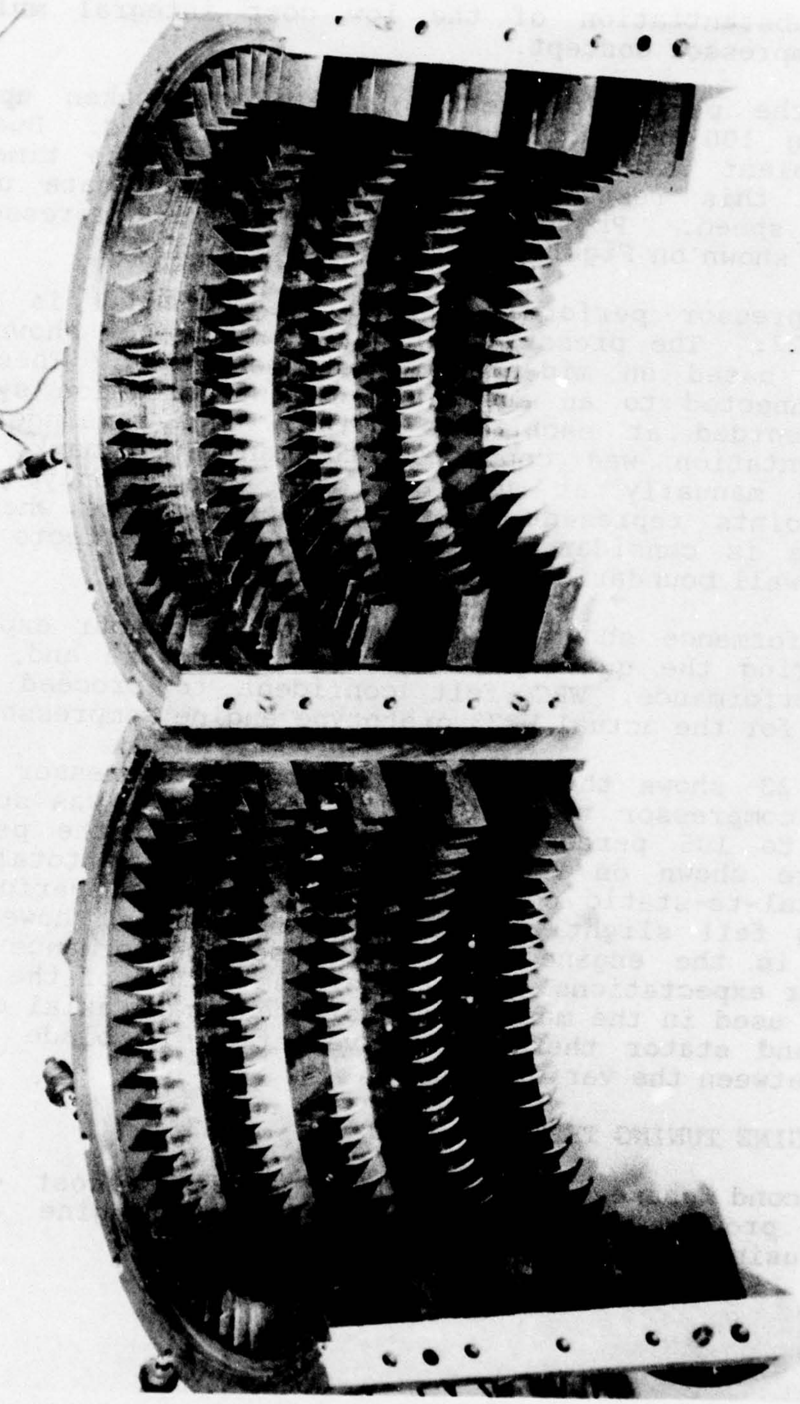


Figure 20. Large Scale Compressor Stator After Test

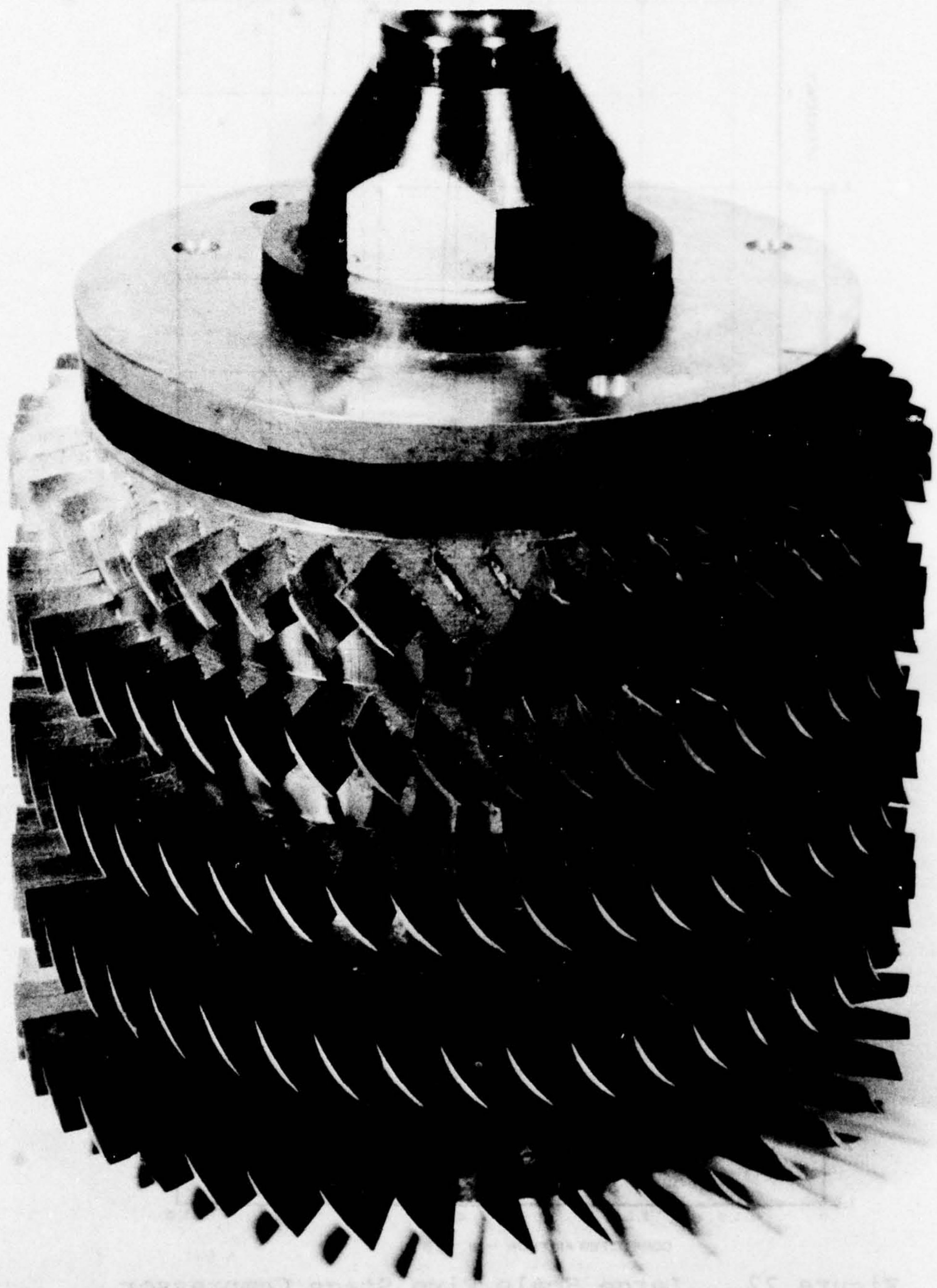


Figure 21. Large Scale Compressor Rotor After Test

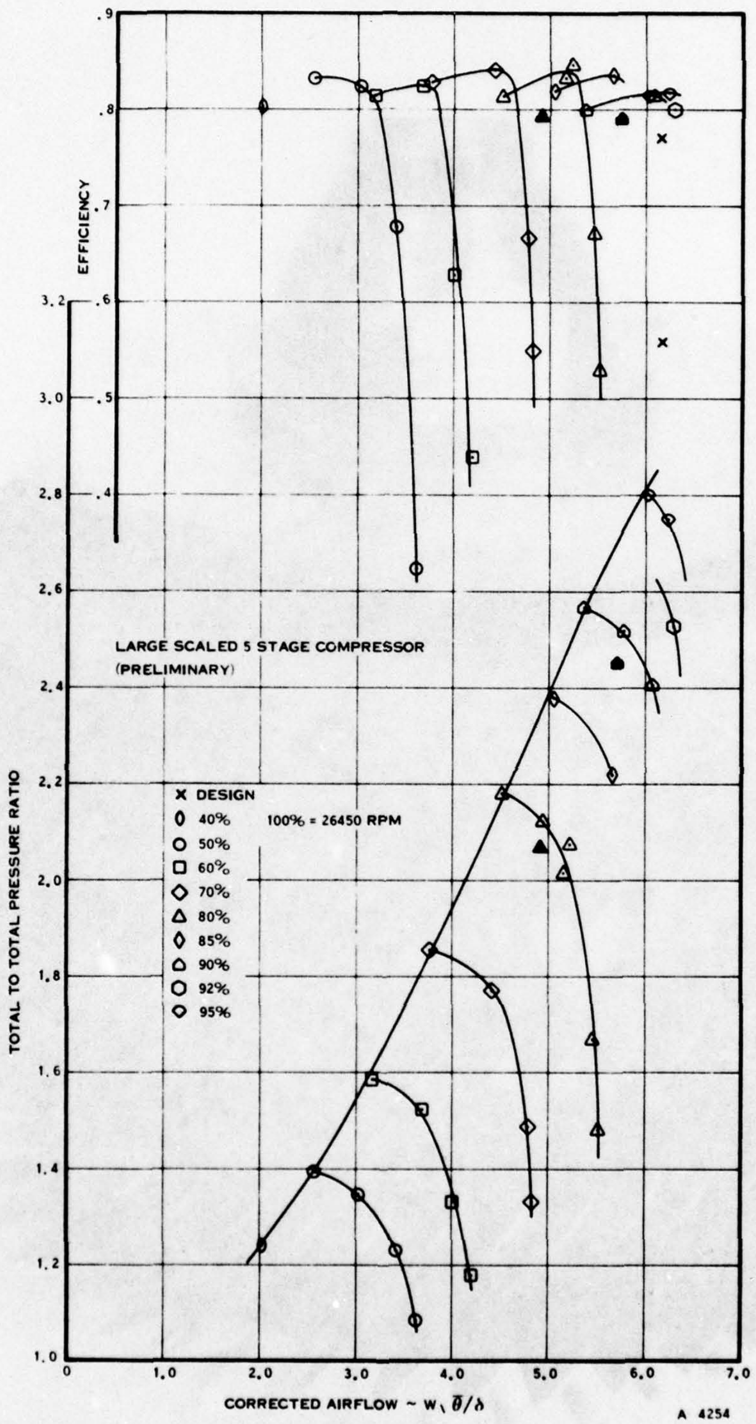


Figure 22. Large Scale Five Stage Compressor Performance Map

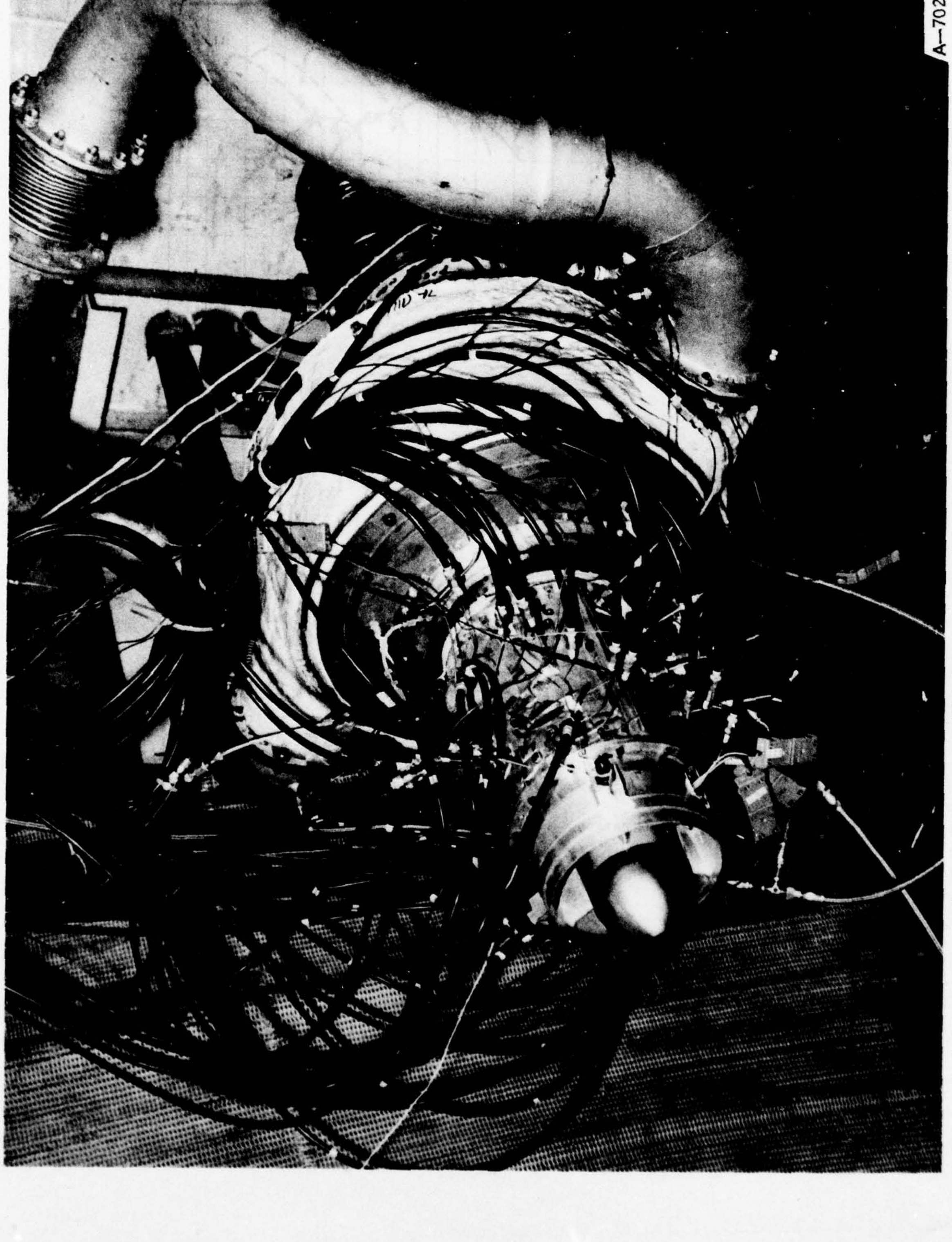


Figure 23. WR33 Compressor Installation on Test Rig

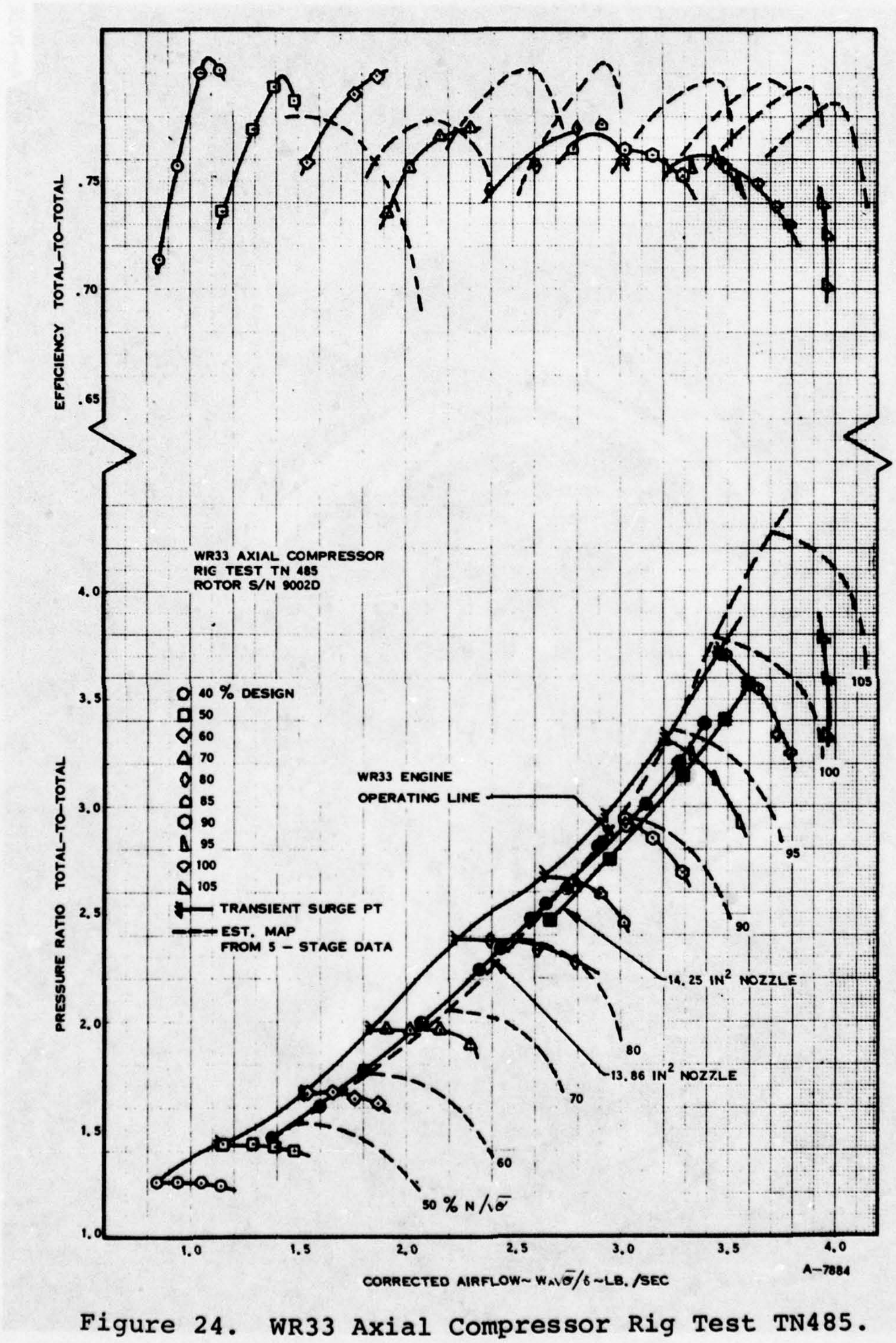


Figure 24. WR33 Axial Compressor Rig Test TN485.

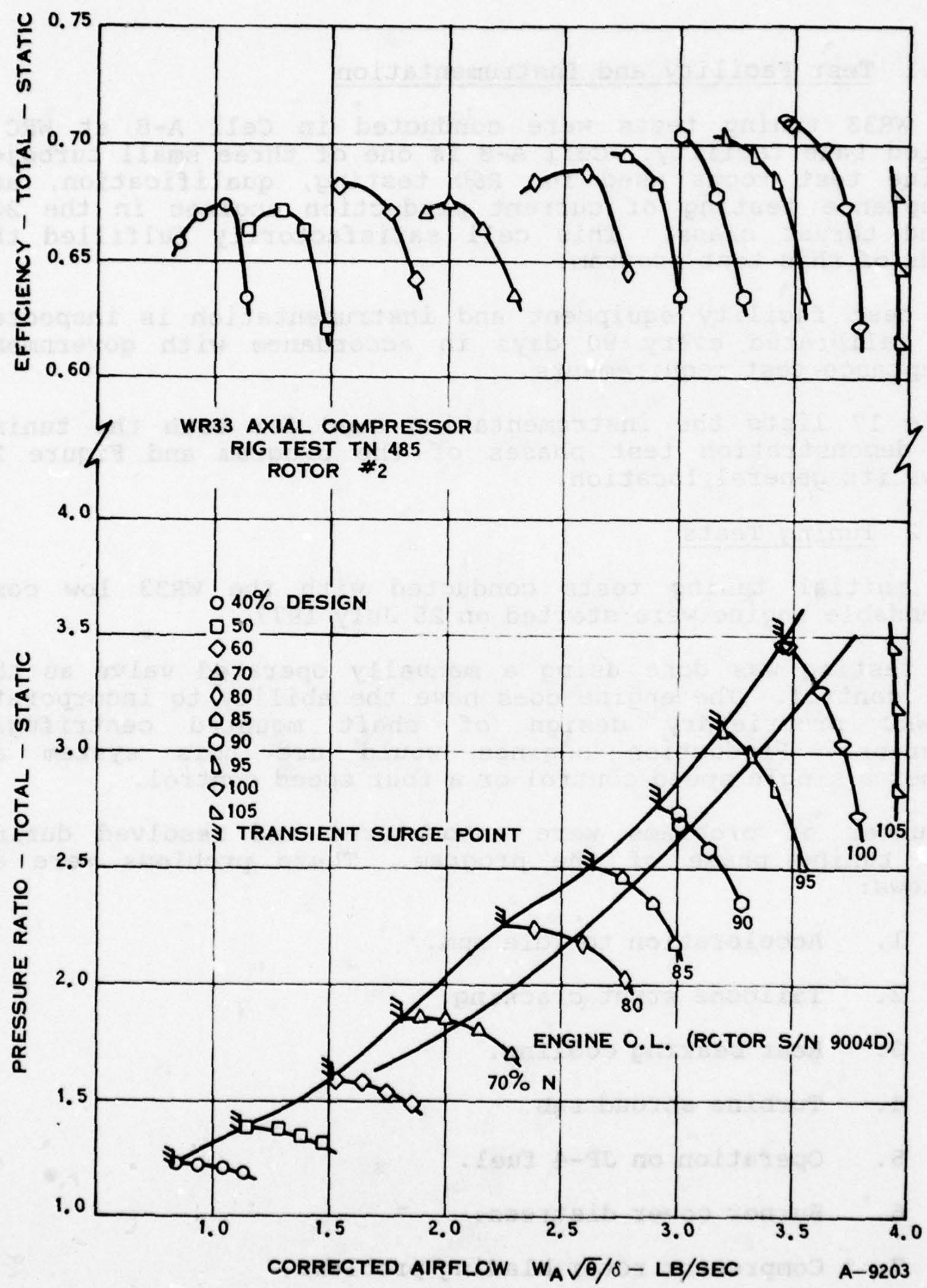


Figure 25. WR33 Six Stage Compressor Performance

3.2.1 Test Facility and Instrumentation

The WR33 tuning tests were conducted in Cell A-8 at WRC's Walled Lake facility. Cell A-8 is one of three small turbojet engine test rooms used for R&D testing, qualification, and acceptance testing of current production engines in the 200 pound thrust class. This cell satisfactorily fulfilled the needs of this test program.

The test facility equipment and instrumentation is inspected and calibrated every 90 days in accordance with government acceptance test requirements.

Table 17 lists the instrumentation used for both the tuning and demonstration test phases of the program and Figure 26 shows its general location.

3.2.2 Tuning Tests

The initial tuning tests conducted with the WR33 low cost expendable engine were started on 25 July 1977.

All testing was done using a manually operated valve as the fuel control. The engine does have the ability to incorporate a WRC proprietary design of shaft mounted centrifugal governor. Production engines would use this system as either a single speed control or a four speed control.

A number of problems were encountered and resolved during this tuning phase of the program. These problems were as follows:

1. Acceleration to idle rpm.
2. Tailcone strut cracking.
3. Rear bearing cooling.
4. Turbine shroud rub.
5. Operation on JP-4 fuel.
6. Burner cover distress.
7. Compressor rotor blading problems.

TABLE 17. INSTRUMENTATION LIST

PARAMETERS	INSTRUMENT	MANUFACTURER	RANGE	WRC S/N
Speed	Frequency Counter	Anadex	99,999 Hz	1597
Fuel Consumption	Flowmeter Flowmeter	Fischer & Porter Fischer & Porter	#/hr #/hr	ICN 1649 ICN 3100
Exhaust gas temperature	Chromel-Alumel Potentiometer	Leeds & Northrop (L&N)	0-2000^{o}F	1290
Thrust	Thrust Stand	Hagan (Wallace & Tiernan Gauge)	0-250#	1440A 1550
Air Inlet Temperature Fuel Temperature Front Bearing Outer Race Temperature Compressor Discharge Temperature	Copper-Constantan Thermocouple Indicator Type T	Howell	0 to 749^{o}F	392
Rear Bearing Outer Race Temperature	Chromel-Alumel Thermocouple Indicator	Doric	-100 to 2400^{o}F	2045
Inlet WP	Manometer	Meriam	0-50" H_2O	ICN 1113
Exhuast Static Pressure	Manometer	Meriam	0-30" Hg Vacuum	1092
Burner Seals Exhaust Total Pressure Compressor Discharge Pressure	Pressure Gage	Wallace & Tiernan	0 to 200" Hg	1550
Vibration Front Rear	Vibration System	General Radio SLM Endevco Pick-up Endevco Pick-up	0-3000 Hz	1555
Fuel Pressure	Pressure Gage	Ashcroft	0-100 psi	2046

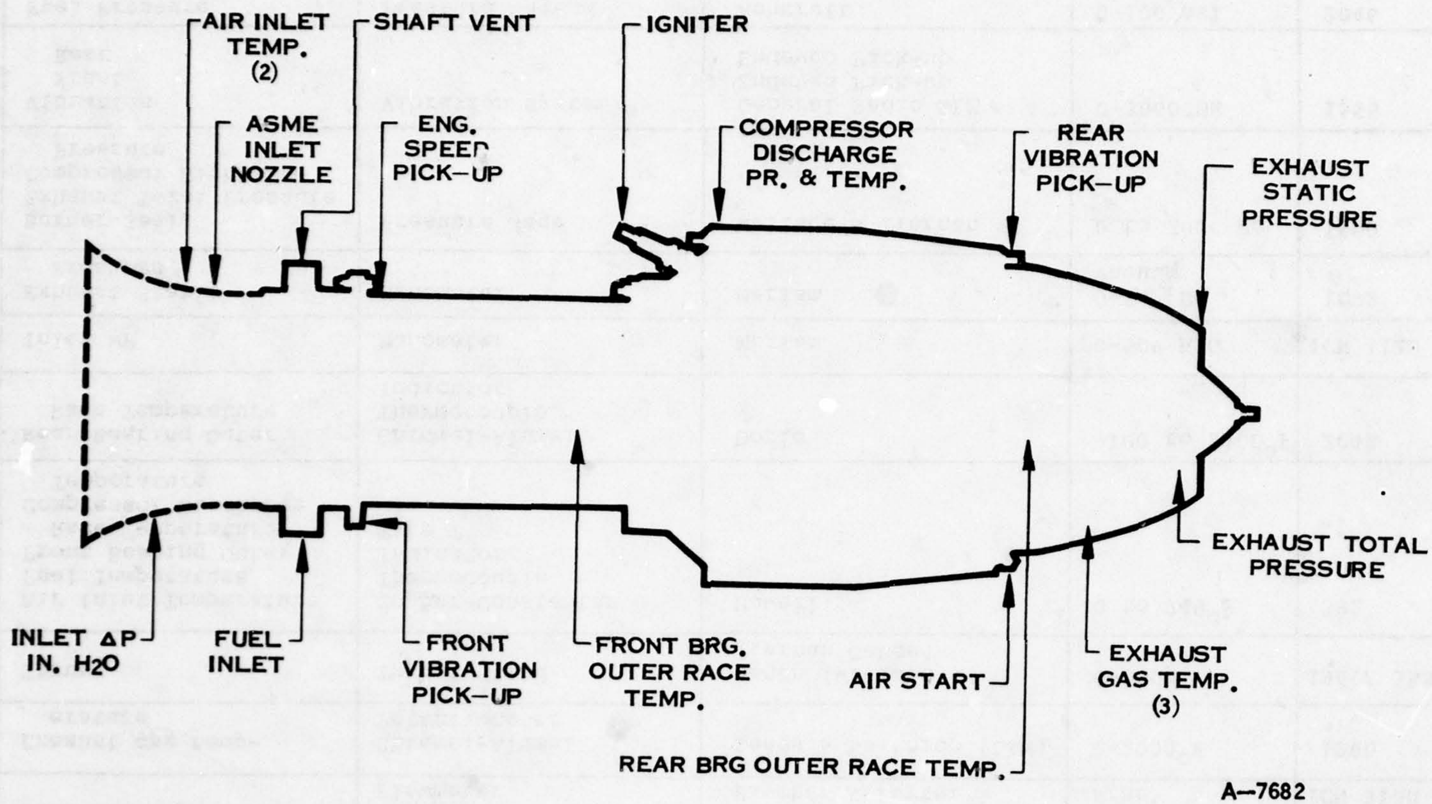


Figure 26. WR33 Engine Instrumentation

The first self-sustaining operation of the WR33 engine took place on 4 August 1977. In order to achieve acceleration up to idle speed, it was necessary to change from air impingement cranking on the compressor rotor to air impingement cranking on the turbine rotor.

Test results indicated that there were two problems associated with the compressor rotor impingement. First, we were unable to crank the rotor to sufficient speed by this method, even though three circumferential impingement locations were used. The optimum air pressure at the impingement nozzles was determined to be approximately 50 psig, which resulted in a cranking speed of 12 percent rpm. Further increases in pressure actually decreased the cranking speed, and at 100 psig, only 9 percent rpm could be obtained.

Secondly, the impingement air choked the last compressor stages causing the early stages to stall, with the net result that the compressor was generating very little pressure rise. Applying air impingement to the turbine rotor was accomplished relatively easily by adding a 3/8 inch diameter tube to the combustor assembly. The inlet end of the tube was aligned with the axis of the engine which allowed it to penetrate the various flanges with a minimum modification to the engine.

Using turbine impingement, we are able to achieve a cranking speed of 17 percent rpm with an air supply pressure of 100 psig.

As the engine speed was increased near rated speed, a sharp rise in vibration was noted at the rear vibration pick-up. After a brief period of running with the rear pick-up indicating levels up to 40 g's, cracks were detected in the tailcone in the vicinity of the struts. Since this sharp rise in vibration was indicative of a resonance condition, it was decided to add doublers to the housing which would effectively increase the stiffness of the structure and raise the resonant frequency above the operating range.

This solution was very effective as shown in Figure 27. Both engines have essentially the same vibration characteristics with neither engine exceeding 15 g's over the operating range. Likewise, the sheet metal cracking in the tailcone area was eliminated.

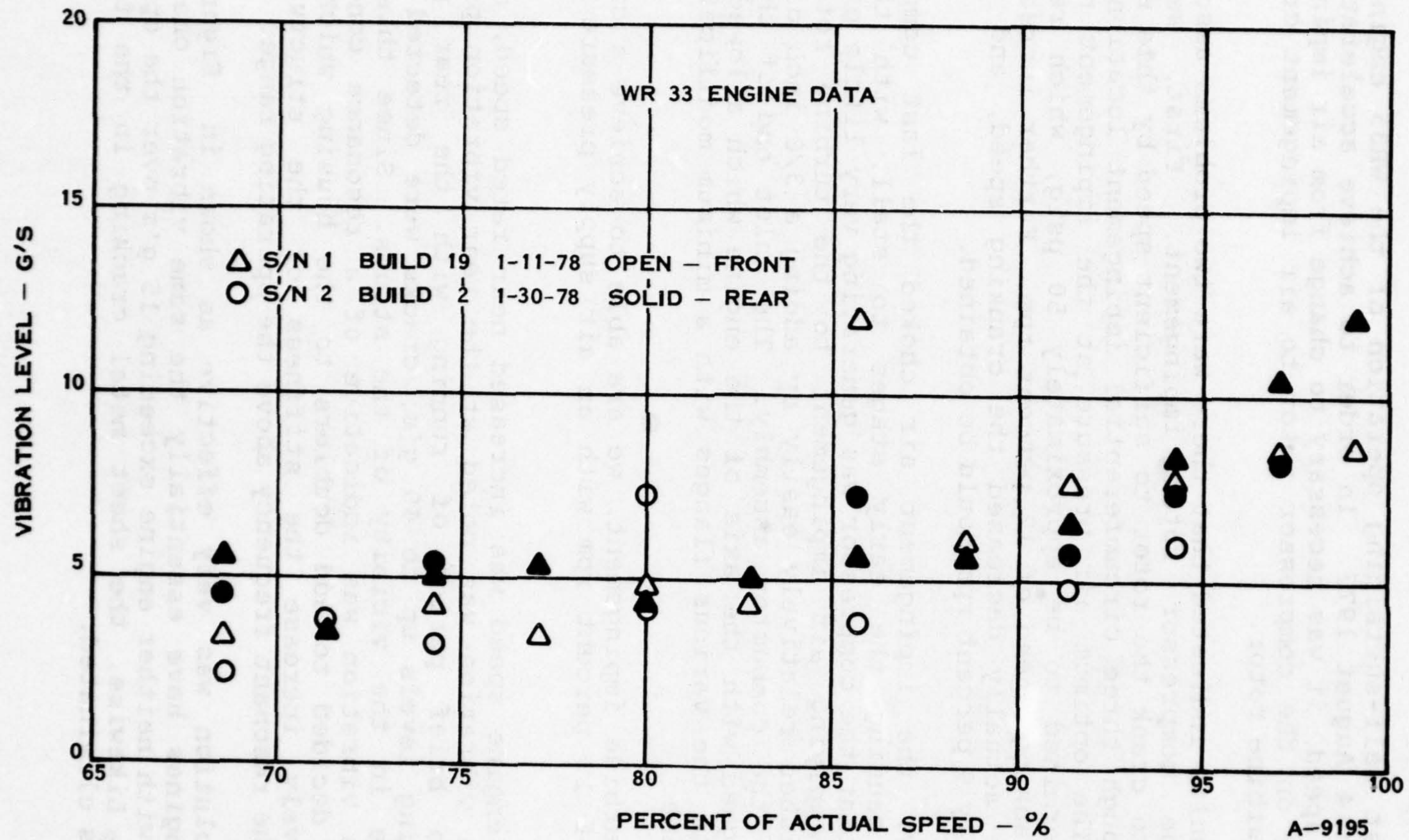


Figure 27. Vibration Level versus Actual Speed

The maximum rear bearing temperature objective was established as 500°F for the grease packed bearing. The initial design intent was to cool the rear bearing area through a series of internal passageways from the compressor, through the rotor hub, the main shaft, across the bearing, and finally out aft of the rear face of the turbine. However, because of problems associated with the early axial compressor rotor castings, it was decided to abandon the air holes through the compressor rotor and to begin engine testing with the rear bearing cooling air supplied from an external source. As engine speed was increased near rated speed, it became necessary to use an excessive amount of air to maintain the temperature limit. At this point, we concluded that it was necessary to reduce the heat flux into the rear bearing area and a Refrasil insulating material was installed in the tailcone area.

This change was very effective and substantially reduced the cooling air flow requirements for the rear bearing. In fact, after insulating the rear bearing cavity, it became practical to supply the cooling air from the engine.

Both engines were equipped with an external cooling air line from the fifth stage axial compressor stator to the rear bearing. The measured bearing temperatures using the self cooled arrangement is shown in Figure 28 for both engines. A slightly larger line was used on engine S/N 1 and it shows a somewhat lower rear bearing temperature, however, both engines operated well below the 500°F limit that was established.

The front bearing runs extremely cool as it is cooled by the inlet air to the engine. There were no bearing failures during the entire tuning and demonstration testing and only one rear bearing was replaced when it was suspected to have become contaminated.

Throughout the tuning period on engine S/N 1, we experienced frequent turbine shroud rubs. These rubs did not occur during engine operation but would occur during rundown and were caused by the thin shroud cooling much more rapidly than the heavier turbine rotor. Although this is not a problem for a single start expendable engine, it was an annoyance during the engine tuning phase.

Engine S/N 2 which had somewhat more radial clearance for the shroud to expand into during operation did not have the shroud rub problem.

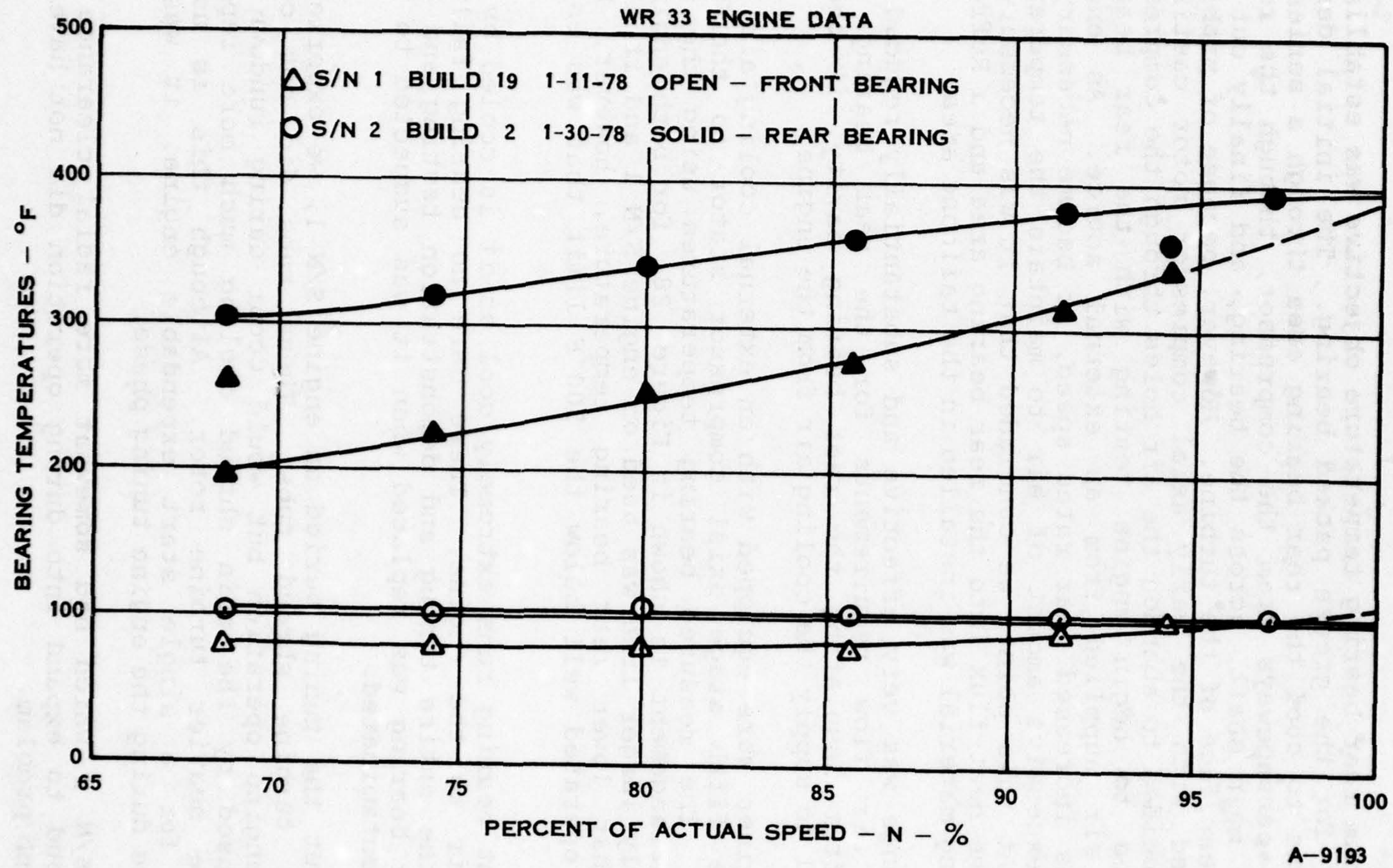


Figure 28. Bearing Temperatures versus Actual Speed

It was found that operation on JP-4 fuel resulted in some fuel being expelled with the air from the shaft vent. The fuel leakage problem was caused by vaporization of the JP-4 fuel within the main shaft during engine operation. Although the engine would operate satisfactorily on JP-4, it was difficult to make an accurate measure of fuel consumption. To eliminate this problem during the tuning and demonstration testing, the fuel was changed to JP-5. It was found that engine starting and combustion performance was equally as good on JP-5 as it was on JP-4.

The combustor operated very well as it was designed, however, the desire to operate the engine for a much longer period than the 30 minute life requirement led to two modifications. The outer wall was modified by forming a corregated section at the aft edge to admit more film cooling along the outer wall. The second change was a material change to the burner cover replacing the 302 stainless steel with Hastelloy X. Measurements also indicated that the combustor pressure drop could be reduced somewhat if an increase in overall engine performance were required.

An inspection of the axial compressor rotor after an engine run at 97 percent rpm revealed that four of the first stage rotor blades were cracked at the trailing edge root. One of the cracked blades was removed for metallurgical examination. The total time on this rotor was 13 hours, 10.5 hours of compressor rig time and 2.5 hours of engine time.

A review of the initial design dynamics analysis indicated that the first bending mode of the first stage rotor blades would be excited by the three inlet struts at 115 percent of rated engine speeds. An actual measurement of the first bending mode of the first stage rotor blades indicates that the three inlet struts excite the rotor in the 68 through 88 percent rpm range. The difference between the calculated and measured values is at least partly attributed to the calculations being based on an integral cast rotor, whereas the measured values are for the bi-cast rotor which has considerably different root attachment stiffness. Likewise, the large difference in frequency from blade to blade is also due to differences in root attachment. In order to place the excitation caused by the inlet struts outside of the engine operating range, the compressor inlet was modified from a three strut inlet to a four strut pattern based on equal spacing of six. This addition changed the engine

speed at which the blades are excited to 44 percent and less, which is below idle and out of the normal operating range. No further difficulty was experienced with the rotor blades after this change was incorporated.

Table 18 presents a chronological test history summary of the engine runs conducted during the tuning phase of the program.

The measured engine performance for engines S/N 1 and S/N 2 at the end of the engine tuning period is shown in Figures 29 through 34. The compressor performance of engine S/N 1 was somewhat better than that of engine S/N 2 and as a result, the thrust was about 5 percent higher for engine S/N 1. All of the other parameters were quite similar between the engines at the rated speed condition.

TABLE 18. CHRONOLOGICAL TEST HISTORY SUMMARY

WR33 Low Cost Expendable Engine

ENGINE S/N 1

Build No.	Date	Comments
1	7/25/77	Initial Build - No Acceleration
2	7/29/77	Modified Rotor Air Start Holes - No Acceleration
3	8/2/77	Modified Rotor Air Start Holes - No Acceleration
4	8/4/77	Turbine Start Impingement - Good Acceleration
5	8/8/77	Reworked Turbine Shroud - Rear Bearing Hot
6	8/16/77	More Stator Clearance - Rub on Engine Rundown
7	8/18/77	New Shim - Rub on Engine Rundown Burner Liner Hot
8	8/24/77	Increase TP Area - Rub on Engine Rundown
9	9/2/77	Fluted Burner - Burner Cool, Rear Bearing Hot
10	9/12/77	Instrumented Rear Bearing - Low ΔP
11	9/27/77	Insulated Rear Bearing - Runs Cool
12	10/13/77	More Air to Rear Bearing - Developed Fuel Leak
13	11/1/77	5GPH Fuel Nozzles - Fuel Leak from Vent
14	11/9/77	Increased Pump Element - Fuel Leak from Vent
15	11/20/77	Add Fuel Spinner - JP-5 - No Fuel Leak w/JP-5
16	11/29/77	New Bladed Rotor - Run to 90% rpm

17	12/12/77	Same Components - Run to 96% rpm
18	12/19/77	Doublers Added to Struts - Run to 99%, 3 Blades Cracked
19	1/13/78	Modified Inlet Struts - Rear Bearing Good
20	1/30/78	Hast X Burner Cover - 200^{O}F EGT Spread
21	2/7/78	Modified Shroud - 1/2 hr Endurance Run
22	DEMONSTRATION RUN	

TOTAL TIME 9:12 hrs & 67 Starts

ENGINE S/N 2

1	1/24/78	Initial Build - Good Run
2	1/26/78	Increased TP Area - Good Run 1:26 Hr
3	2/7/78	Demonstration Test - Good Run Cal.

TOTAL TIME: 3:06 hrs & 29 starts

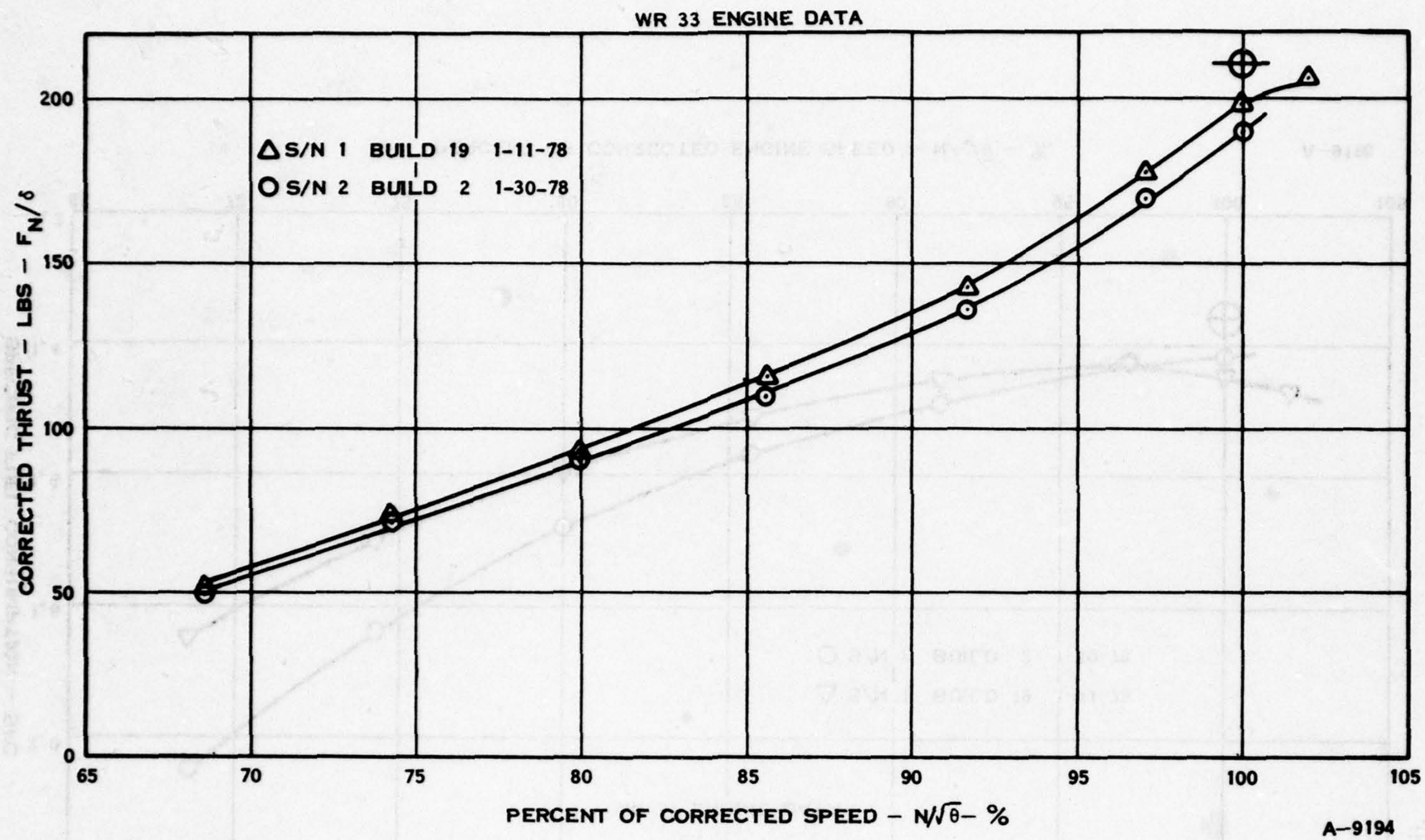


Figure 29. Corrected Thrust versus Corrected Speed

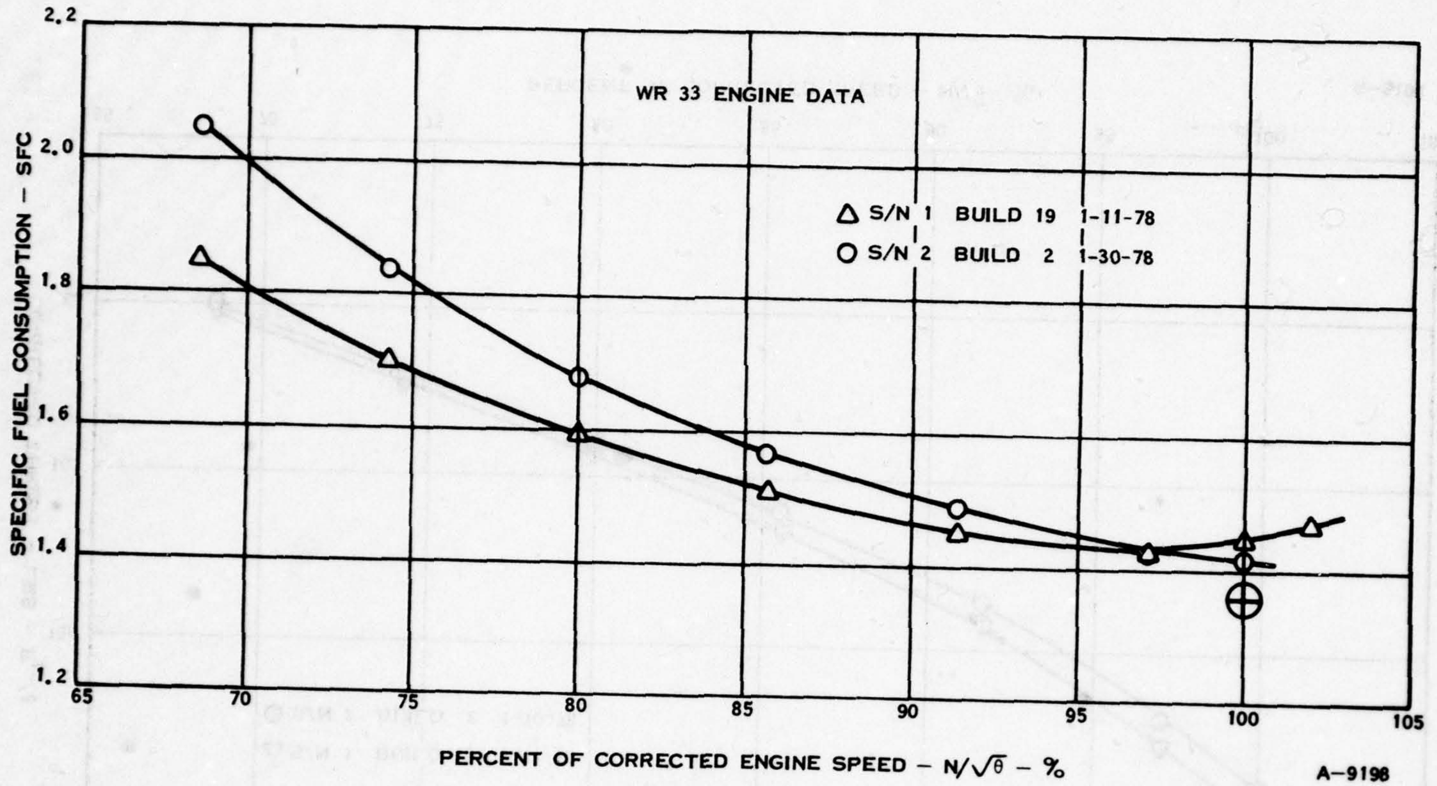


Figure 30. Specific Fuel Consumption versus Corrected Speed

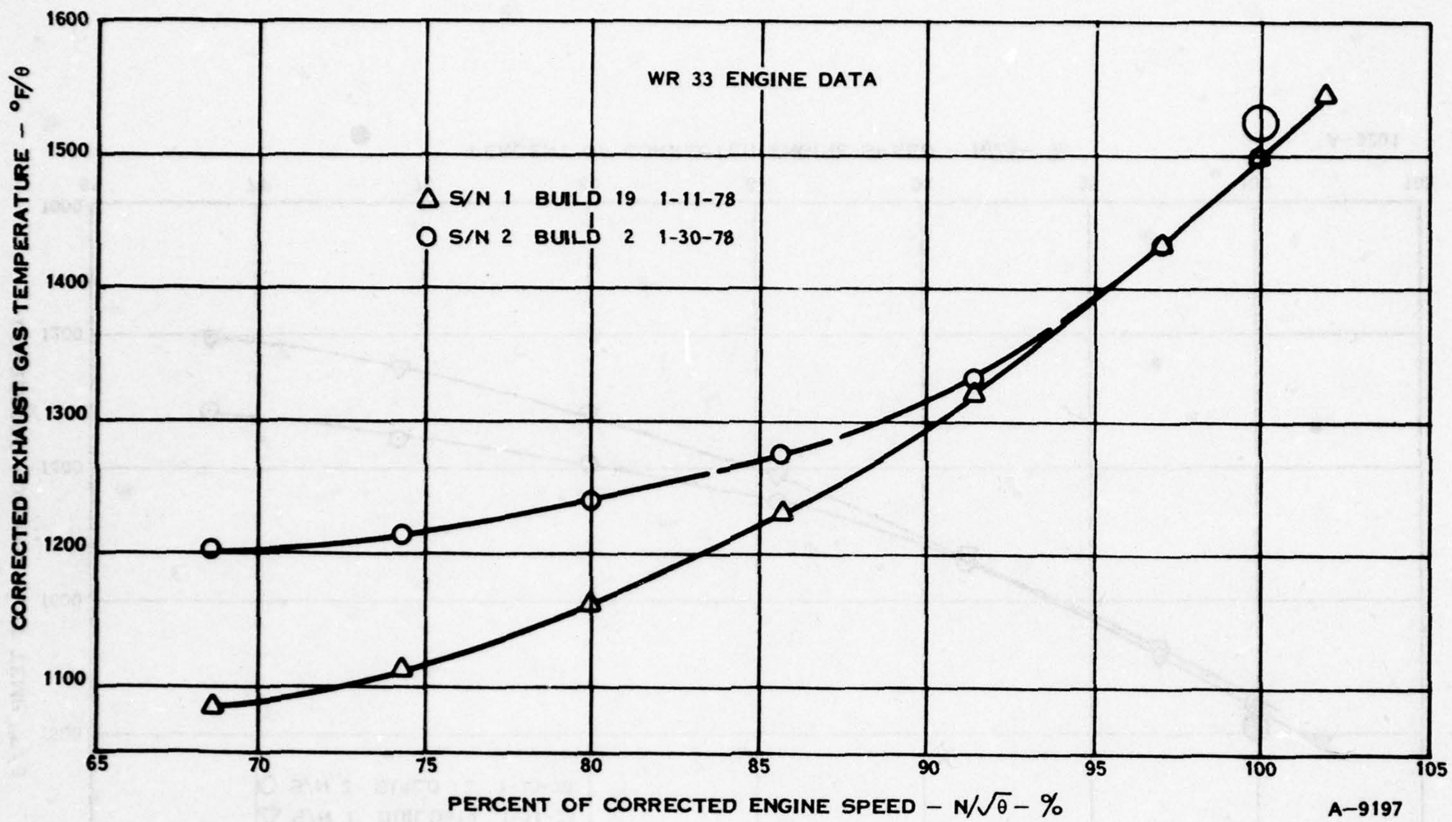


Figure 31. Corrected Exhaust Gas Temperature versus Corrected Speed

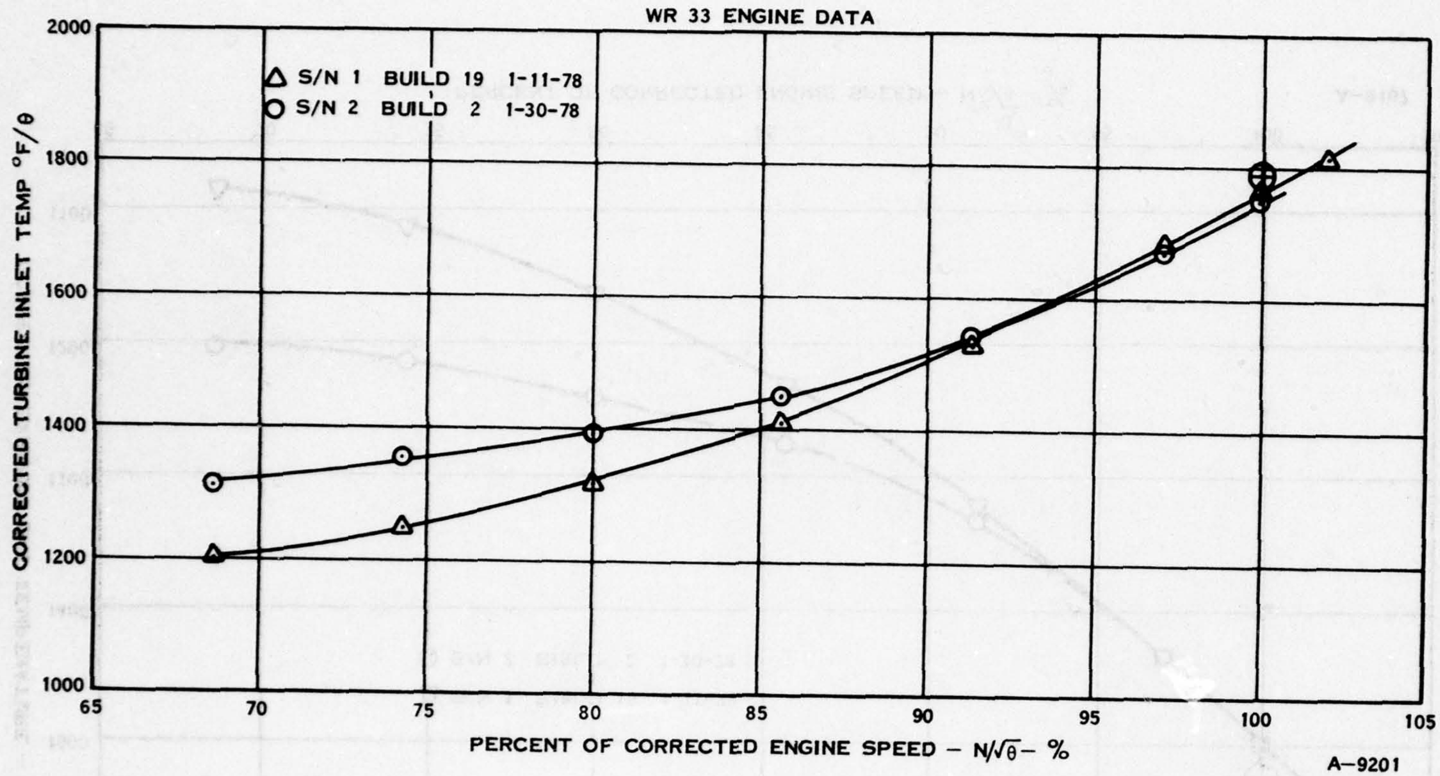


Figure 32. Corrected Inlet Temperature versus Corrected Speed

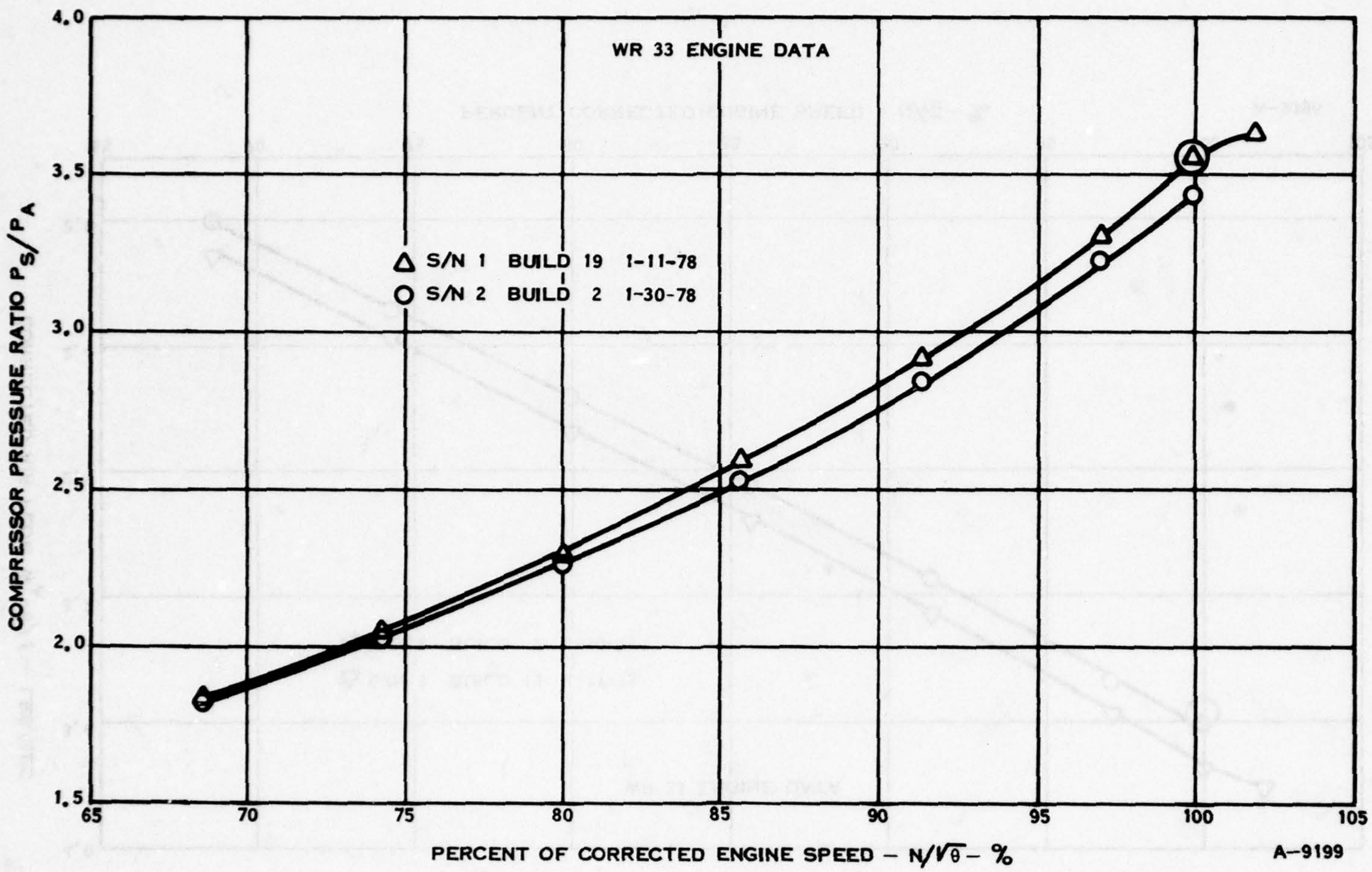


Figure 33. Compressor Pressure Ratio versus Corrected Speed

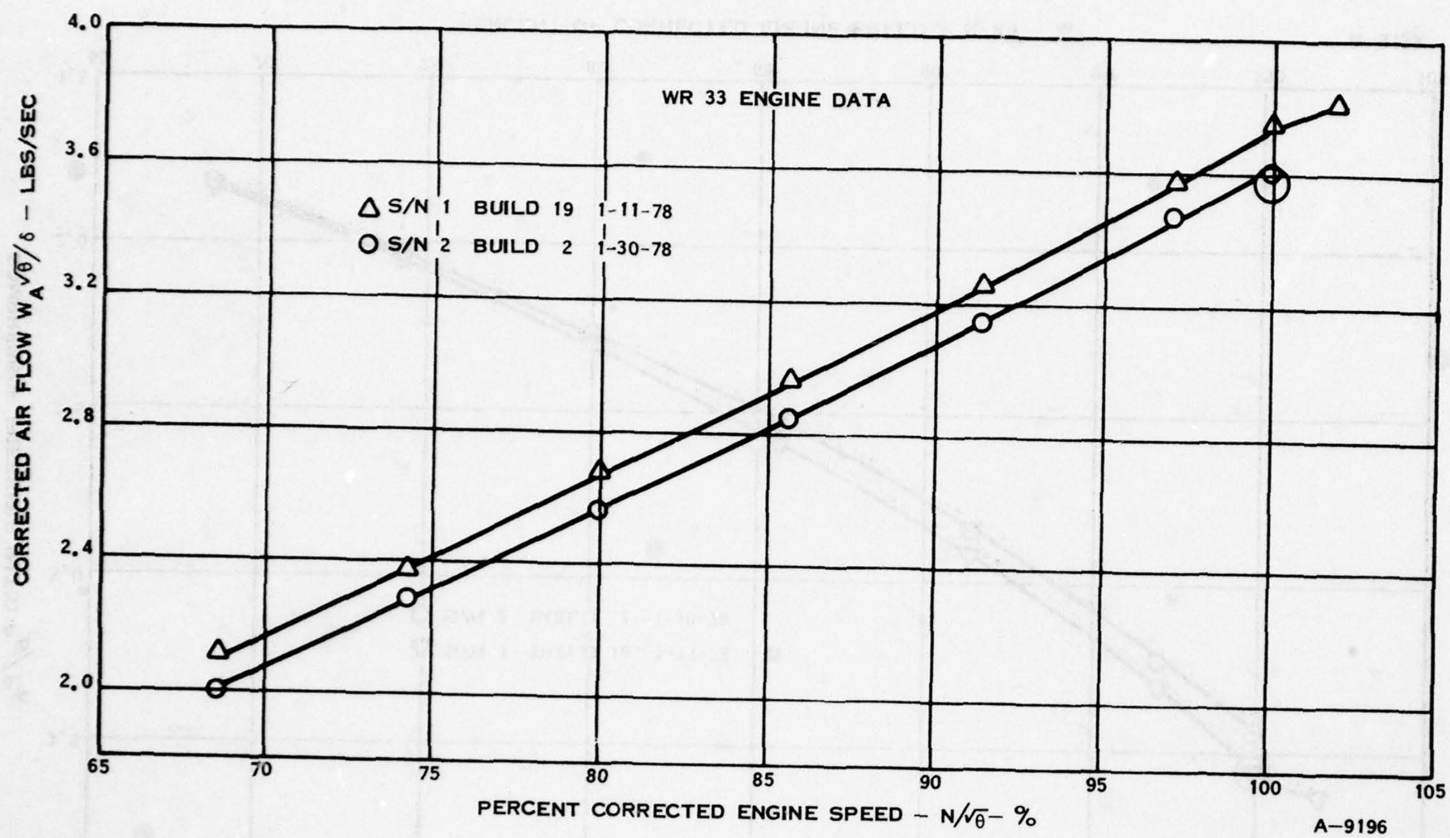


Figure 34. Corrected Air Flow versus Corrected Speed

SECTION 4

TASK III DEMONSTRATION TESTS

Task III of the low cost expendable engine program required a thirty minute demonstration run at 100 percent engine speed and Mach 0.7.

The use of the no-flow ejector which is rigidly attached to the engine precluded the use of the standard thrust stand for measuring thrust. As a result, a total pressure rake at the engine jet nozzle exit along with three static pressure probes at the exit were used to calculate the thrust under the simulated ram condition. The engine was first operated on the standard thrust stand without the ejector but with the instrumentation installed in order to obtain an effective jet nozzle flow coefficient. Using this approach, the estimated accuracy of the gross thrust measurement is ±2 percent and the net thrust estimate is accurate within ±5 percent. The accuracy of the thrust measurement under static sea level conditions is within ±1 percent.

Because of the difficulty of accurately assessing the engine performance under the simulated flight condition when using a no-flow ejector, we elected to demonstrate performance and Mach 0.7 operation separately by running a 30 minute demonstration test on each engine in accordance with the following plan:

Engine S/N 1 - Sea Level Static Test

- Record performance data at the following approximate corrected speeds.

  57 percent
  71 percent
  86 percent
  94 percent
  100 percent

- Run 30 minutes at a corrected thrust in excess of 175 pounds.

Engine S/N 2 - Simulated Flight Test

- Record performance data at the following approximate corrected speeds.

86 percent
94 percent
100 percent

- Run 30 minutes at a corrected speed of approximately 100 percent.

4.1 TEST FACILITY AND INSTRUMENTATION

The WR33 demonstration test was conducted in Cell A-8 at WRC's Walled Lake facility. Cell A-8 is one of three small turbojet engine test rooms used for R&D testing, qualification, and acceptance testing of current production engines in the 200 pound thrust class. This cell satisfactorily fulfilled the needs of this test program.

The test instrumentation included that necessary to obtain the following data:

Net thrust - pounds
Engine speed - rpm
Turbine outlet temperature - ^{o}F
Engine inlet air flow - pounds per second
Fuel flow - pound per hour
Compressor discharge pressure - in. Hg. gauge
Compressor discharge temperature - ^{o}F
Engine vibration level - g's
Front and rear bearing outer race temperature - ^{o}F
Inlet air temperature - ^{o}F
Barometric pressure - in. Hg.
Exhaust ejector static pressure - in. Hg
Exhaust stream total pressure - in. Hg.

In addition, the following parameters were recorded for each test.

Tailpipe ID - Inches
Start No. this build
Start time

The sea level static engine data was reduced on the WRC Turbojet Data Reduction Program. This program calculates overall engine performance from the measured data.

The output data from the program include the following parameters corrected to standard temperature and pressure:

rpm - corrected speed
SFC - specific fuel consumption
T4 - corrected exhaust gas temperature
F - corrected thrust
Δ - standard pressure correction factor
θ - standard temperature correction factor
√θ - square root of standard temperature correction factor
Wf - correct fuel flow
P21 - compressor pressure ratio
Airfl - corrected air flow
TIT - turbine inlet temperature

The performance under the simulated ram conditions was calculated based on the measured air flow, exhaust pressure and temperature, and the static pressure at the nozzle exit as described in Paragraph 4.4.

4.2 DEMONSTRATION TESTS

The demonstration tests, witnessed by AFAPL personnel, were successfully completed on 7 February 1978. Both engines were used for the test so that sea level as well as simulated flight performance could be demonstrated.

Testing on engine S/N 1 consisted of a calibration run between corrected speeds of 57 and 100 percent rpm followed by a thirty minute sustained run at a thrust level in excess of 175 pounds. A summary of these calibration data completed with engine S/N 1 is presented in Table 19. This unit was operated for a total of fifty eight minutes during the demonstration test.

Using engine S/N 2, the 0.7 Mach Number conditions were simulated by the use of a no flow ejector as shown in Figure 35. The ram pressure ratio associated with Mach 0.7 is 1.39 and was simulated by adjusting the position of the ejector. This test generated representative flight conditions and jet nozzle pressure ratios. The inlet stagnation pressure and temperature were the prevailing ambient values at the time of the test. This setup is shown in Figures 36 and 37.

Engine S/N 2 was used for the simulated flight test, which consisted of a calibration run at approximately 85, 94 and 100 percent engine speeds followed by a sustained thirty minute run at a corrected speed of 100 percent rpm. all under simulated flight conditions. A summary of the test data for engine S/N 2 is shown on Table 19. Engine S/N 2 was operated for a total of 51 minutes during the demonstration test.

TABLE 18. **ENGINE S/N 1 SUMMARY**

Thirty minutes Endurance Test Data - Engine S/N 1:

$\%N/\sqrt{\theta}$	Fn/δ	SFC	EGT/θ
98.0	179.8	1.403	1449
97.8	179.5	1.380	1447
97.8	180.3	1.373	1454
98.0	180.6	1.373	1451
98.0	180.1	1.375	1449
98.0	180.3	1.375	1449
98.0	180.3	1.371	1448

Pre-endurance Calibration Data-Engine S/N 1

$\% N/\sqrt{\theta}$	Fn/δ	SFC	EGT/θ
57.3	36.6	2.446	1205
71.7	66.4	1.824	1188
85.7	114.7	1.482	1238
94.5	160.4	1.391	1387
100.0	195.4	1.383	1513

Post-endurance Calibration Data - Engine S/N 1

$\%N/\sqrt{\theta}$	Fn/δ	SFC	EGT/θ
94.6	159.1	1.386	1364
86.1	115.8	1.466	1232
72.2	68.2	1.774	1157
57.7	37.7	2.403	1189

Note: Data corrected to sea level static conditions.

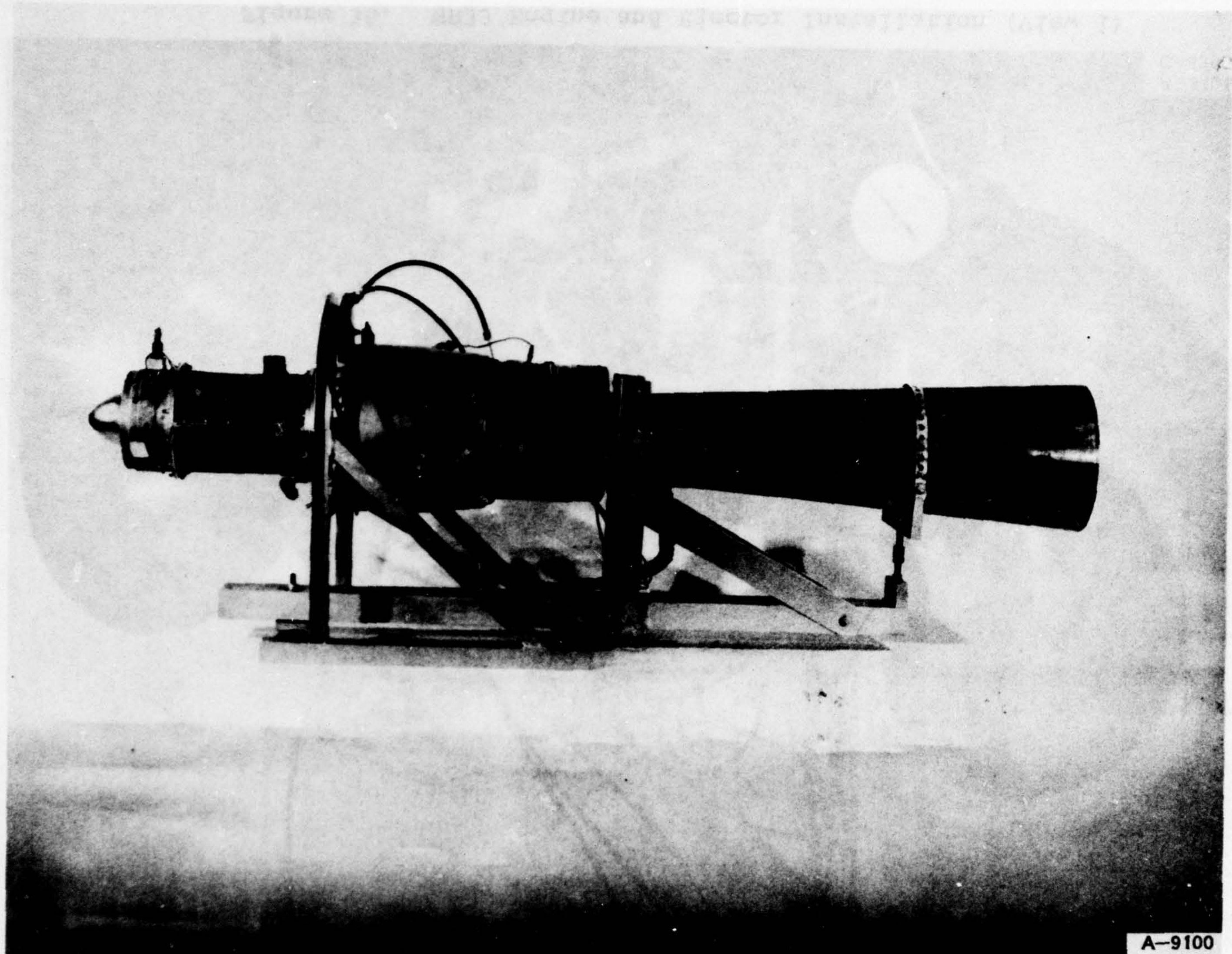


Figure 35. WR33 Engine and Ejector for Altitude Simulation

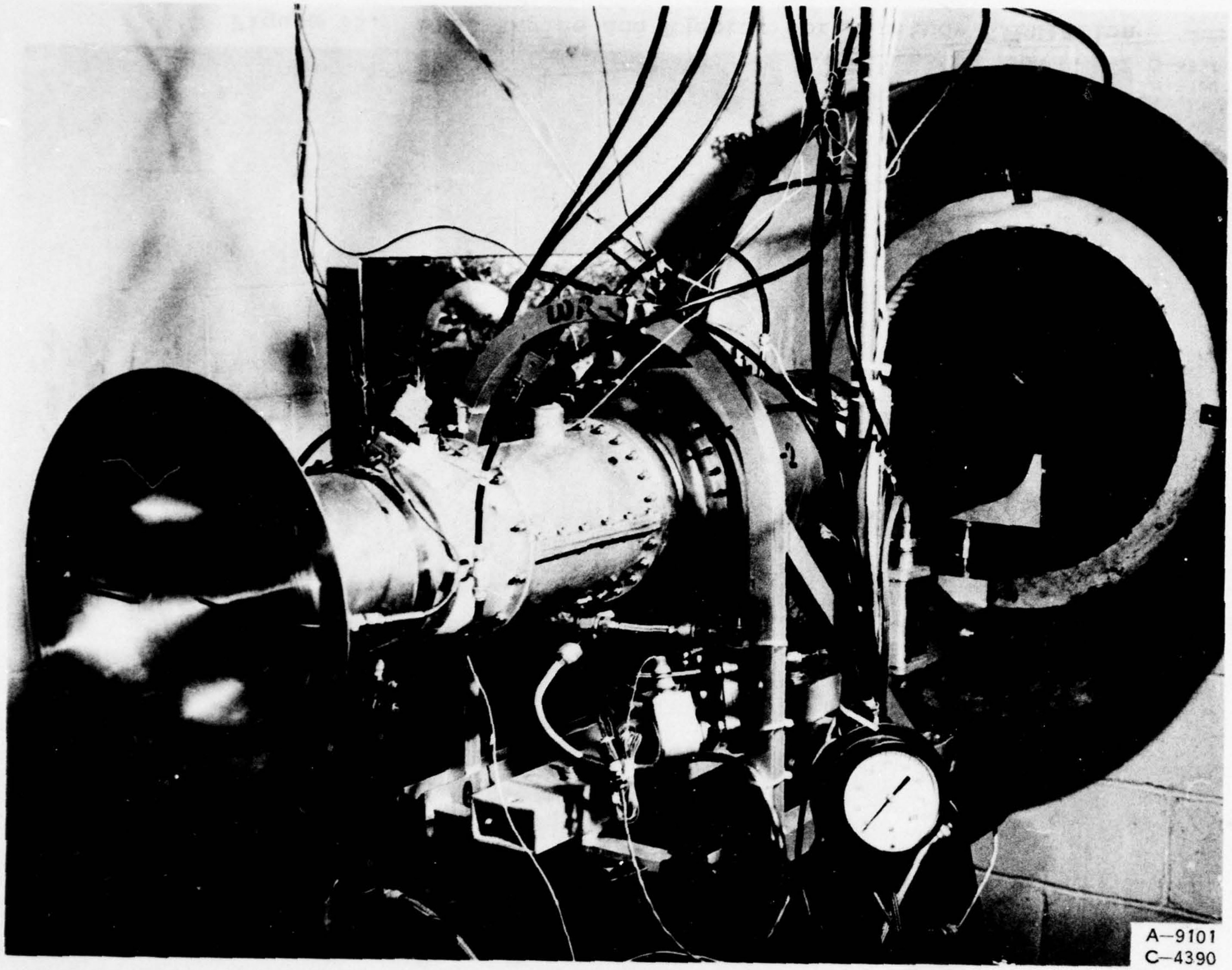
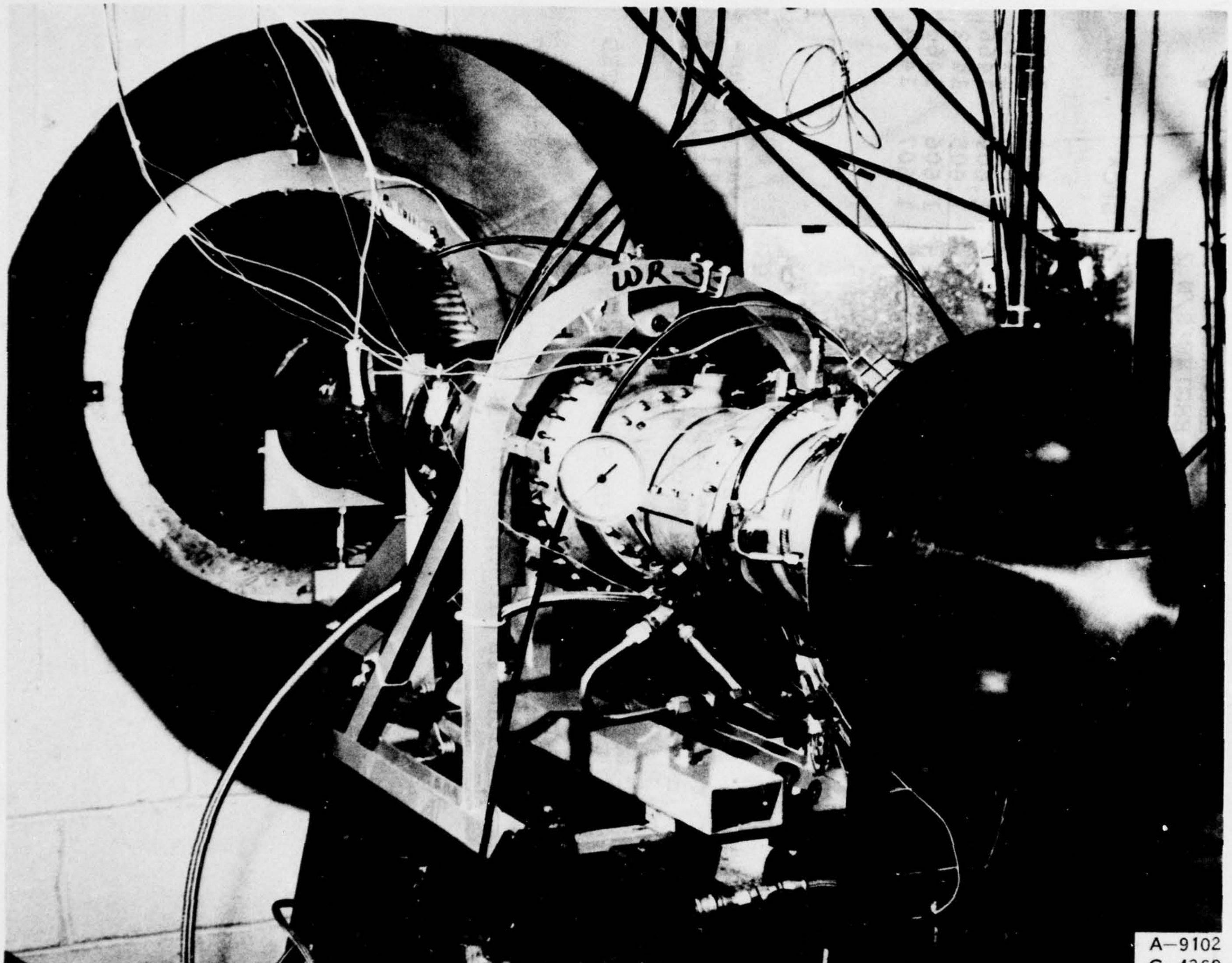


Figure 36. WR33 Engine and Ejector Installation (View 1)



A—9102
C—4369

Figure 37. WR33 Engine and Ejector Installation (View 2)

TABLE 19. THIRTY MINUTE ENDURANCE TEST AT MACH 0.7 - ENGINE S/N 2

SIMULATED ALTITUDE-FT	SIMULATED MACH NO.	$\%n/\sqrt{\theta}$	F_{Gross}	F_{Net}	SFC*	EGT/θ
11,000	0.77	100.6	246	155.9	1.628	1469
11,000	0.77	100.7	246.6	156.3	1.615	1475
11,000	0.79	100.8	248.7	156.3	1.609	1466
11,000	0.79	100.9	249.1	156.6	1.605	1472
11,000	0.79	100.8	248.7	156.5	1.606	1466
11,000	0.79	100.9	248.8	156.2	1.607	1464

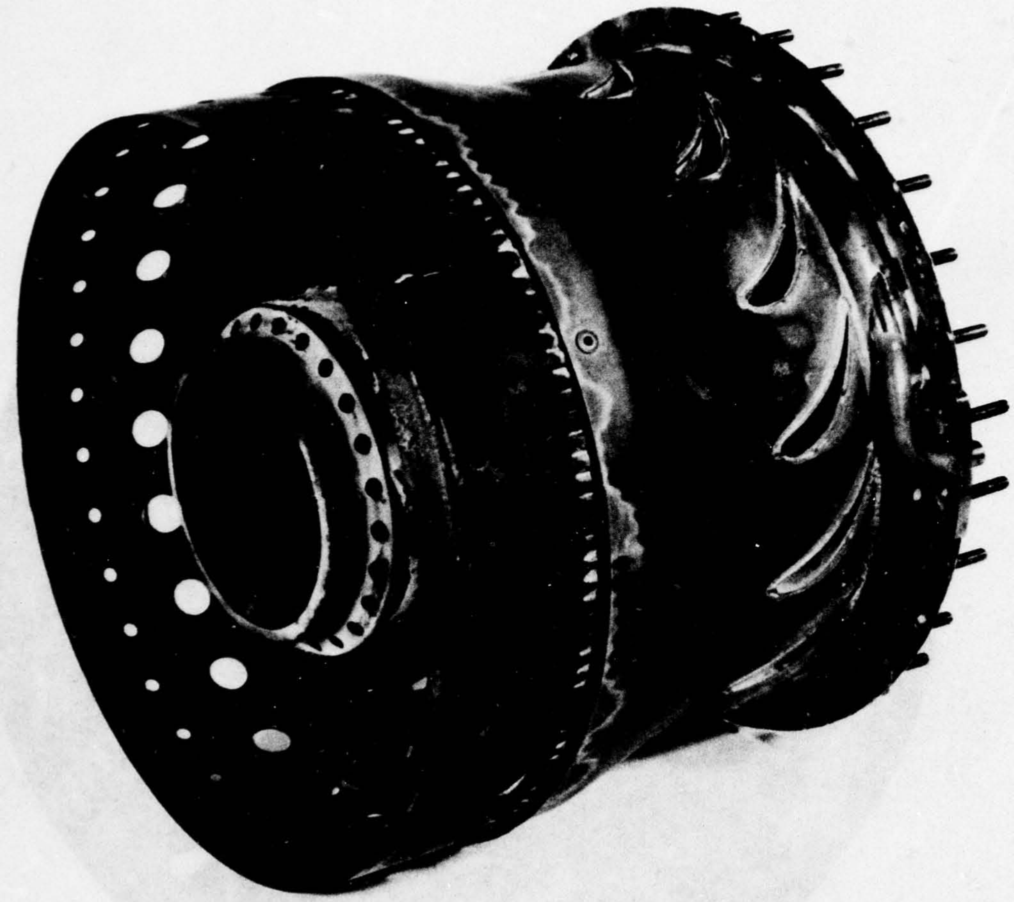
*SFC based on net thrust

4.3 DISASSEMBLY INSPECTION

Following the demonstration test, engine S/N 1 was disassembled for review by the AFAPC representatives. All of the parts were in excellent condition and are shown in Figures 38 through 50.

4.4 ENGINE DATA REDUCTION UNDER MACH NUMBER SIMULATED CONDITIONS

The method used to compute the performance under Mach 0.7 simulated conditions is presented in Appendix B.



A—9103
C—4416

Figure 38. WR33 Combustor Assembly - P/N 100 (View 1)

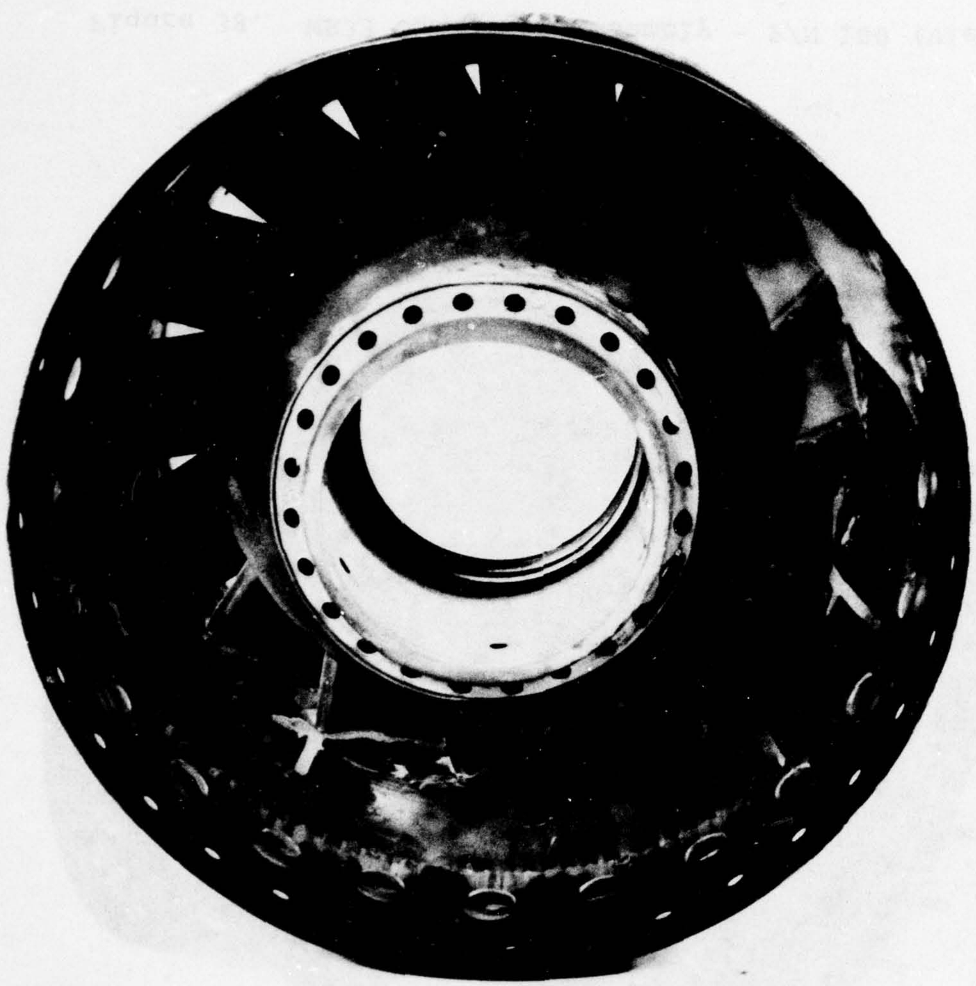
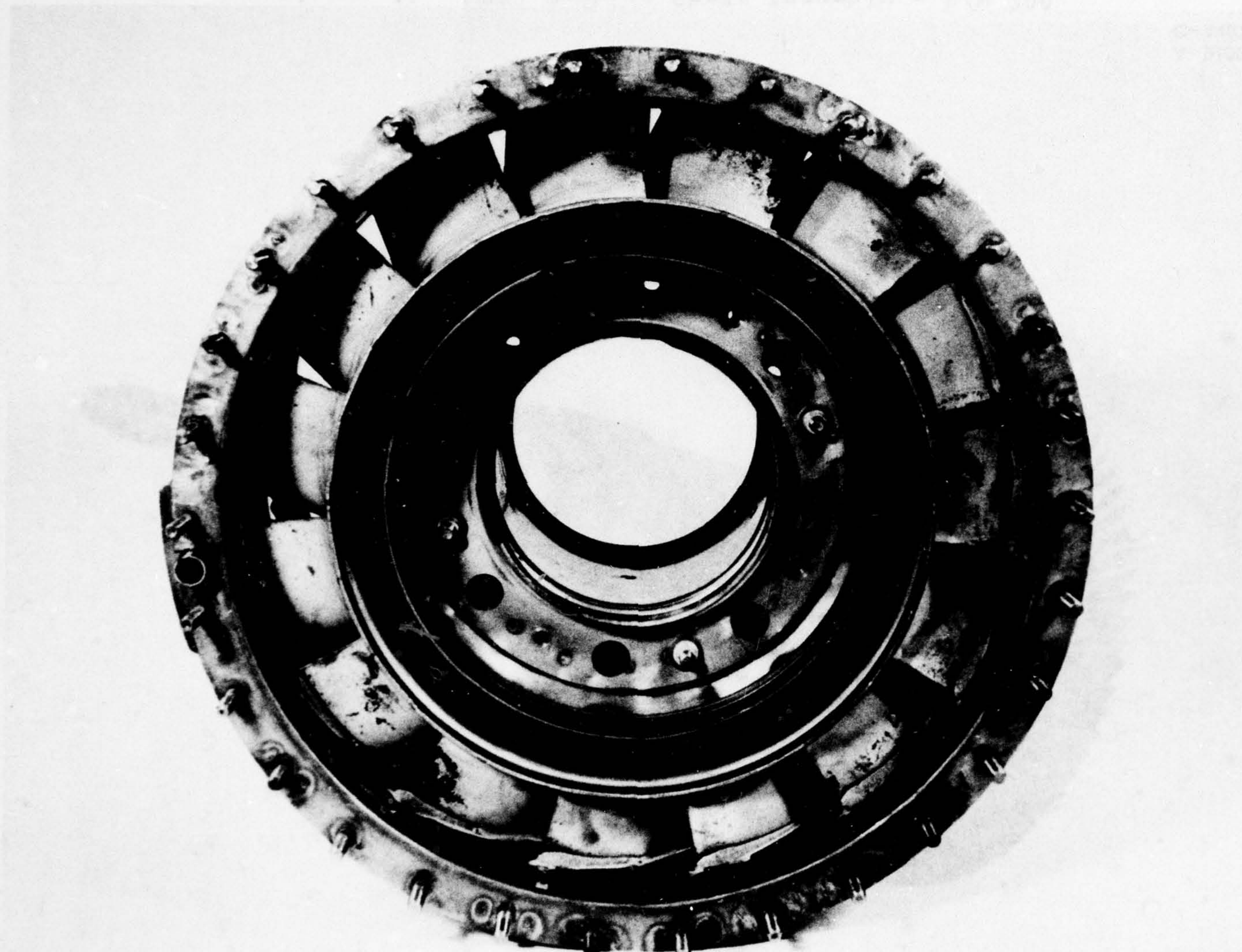


Figure 39. WR33 Combustor Assembly - P/N 100 (View 2)



A–9105
C–4418

Figure 40. WR33 Combustor Assembly - P/N 100 (View 3)

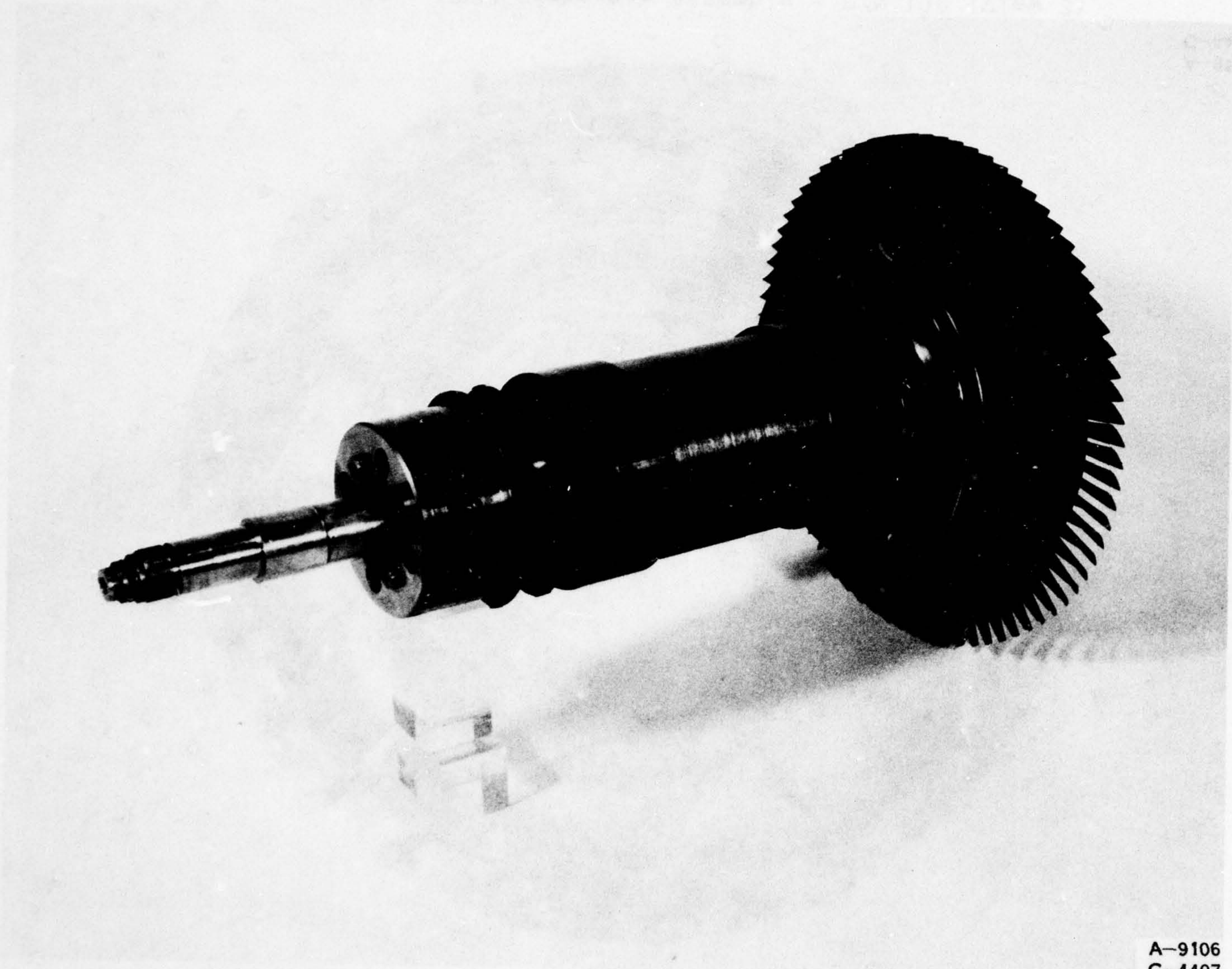
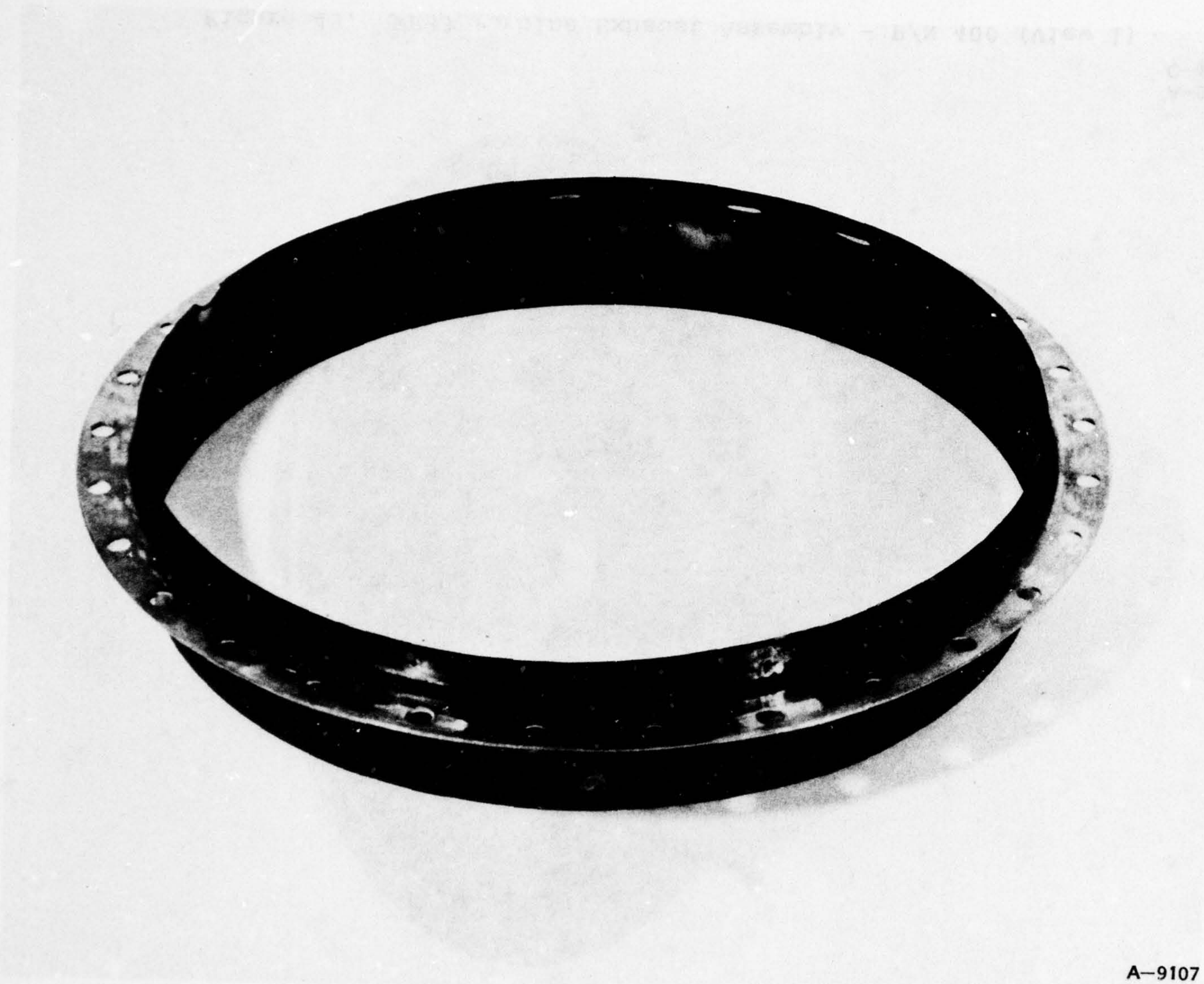
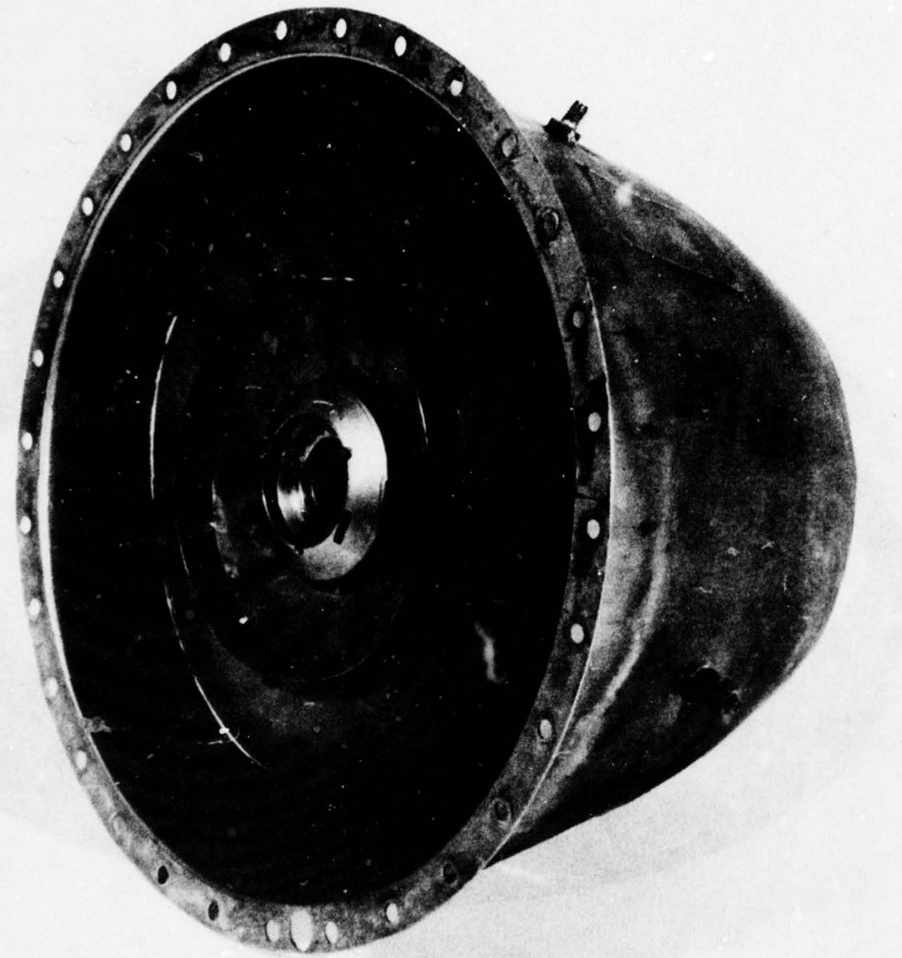

A-9106
C-4407

Figure 41. WR33 Turbine Shaft Assembly - P/N 200



A–9107
C–4409

Figure 42. WR33 Turbine Shroud Assembly - P/N 303



A–9108
C–4410

Figure 43. WR33 Turbine Exhaust Assembly - P/N 400 (View 1)

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LOW COST EXPENDABLE ENGINE.(U)
MAR 78 C A HUBEN, B L METSKER F33615-76-C-2123

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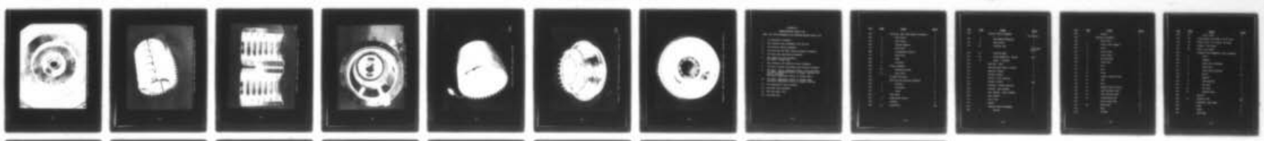

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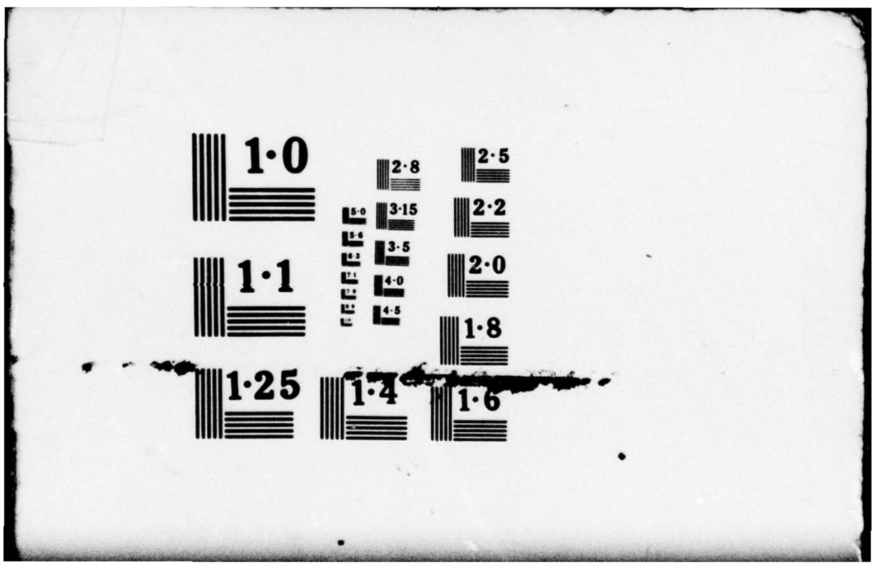
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2·8
2·5
3·15
2·2
3·5
2·0
1·1
4·0
4·5
1·8
1·25
1·4
1·6

Figure 44. WR33 Turbine Exhaust Assembly - P/N 400 (View 2)

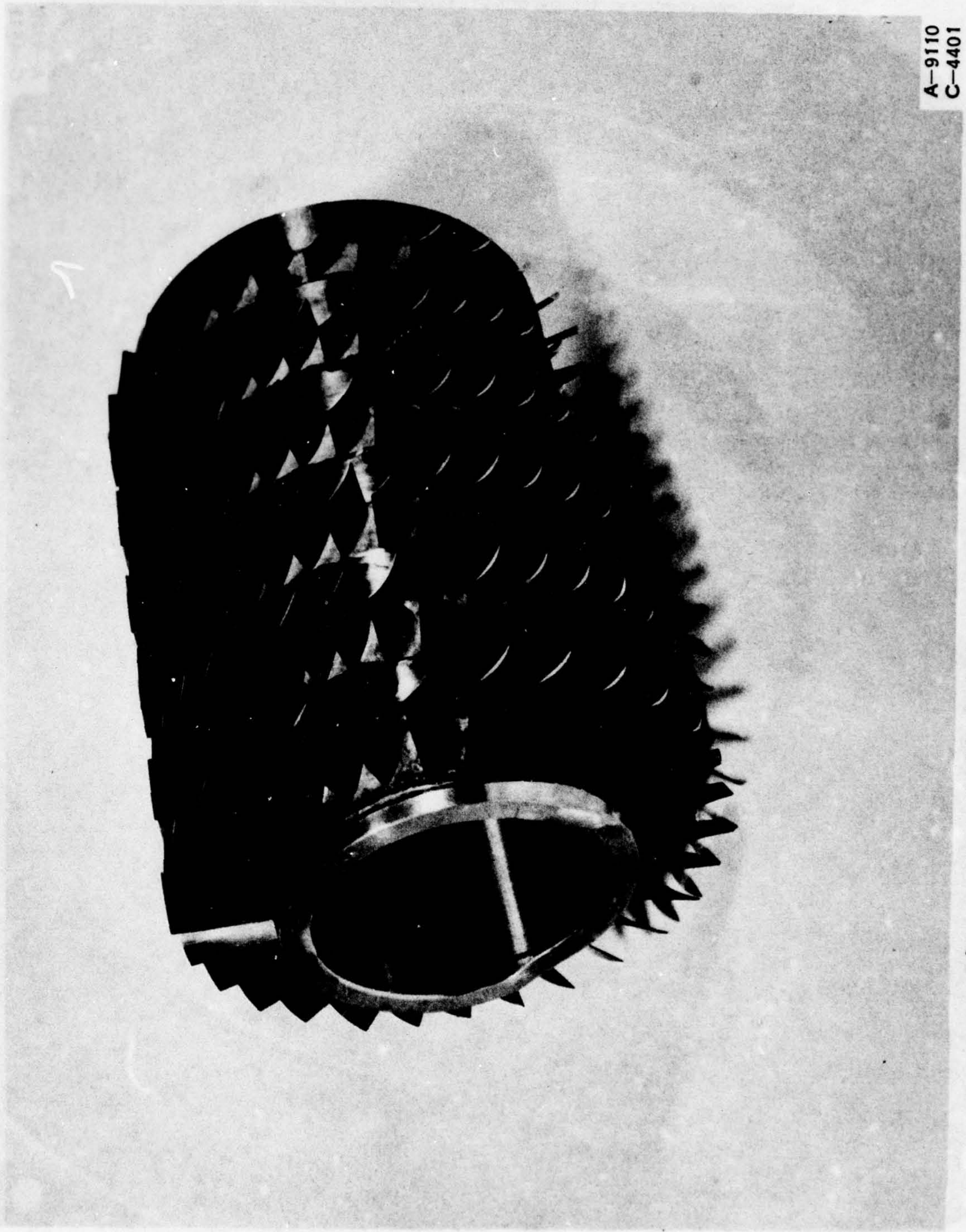


Figure 45. WR33 Axial Compressor Rotor - P/N 723

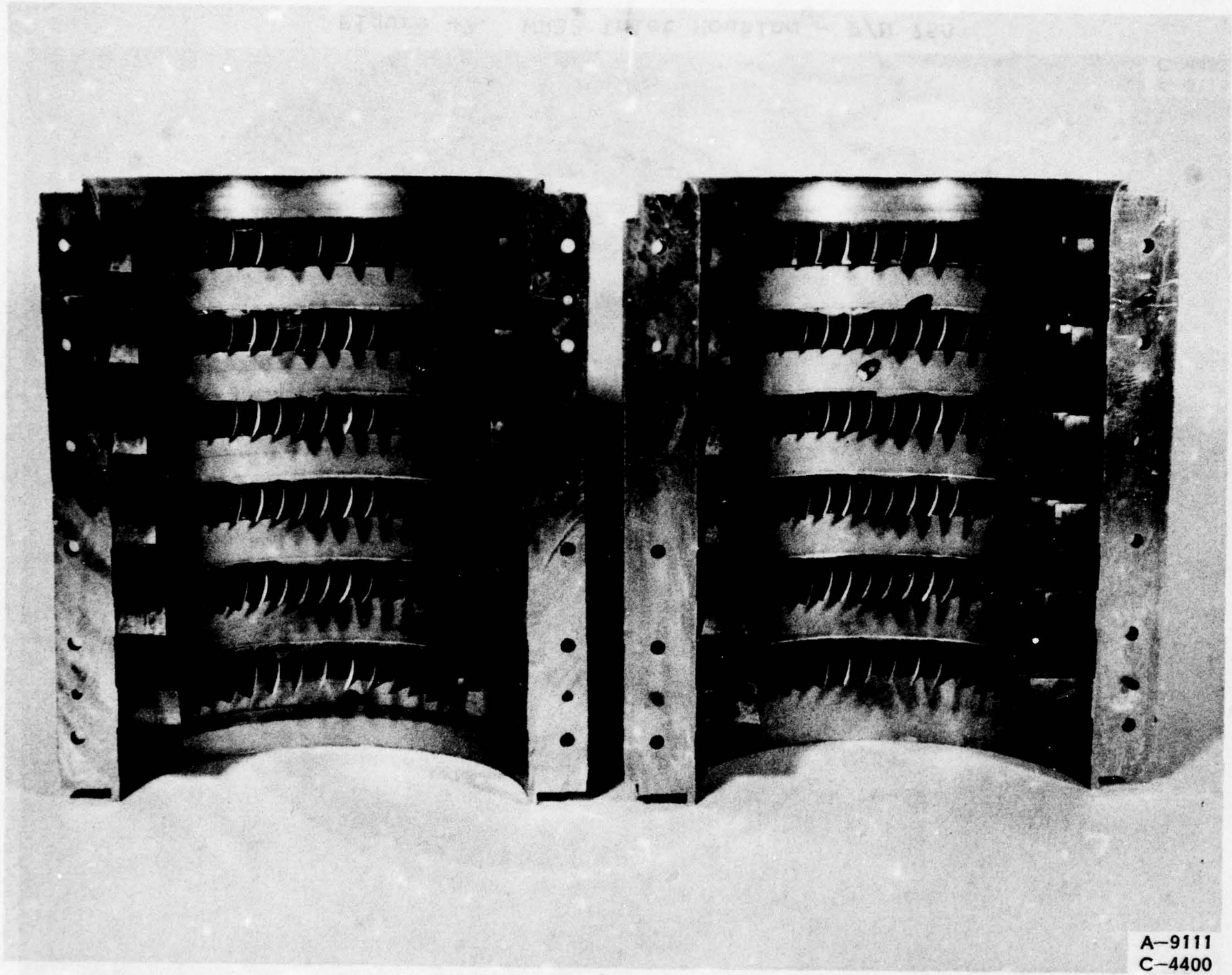


Figure 46. WR33 Axial Compressor Stator - P/N 725

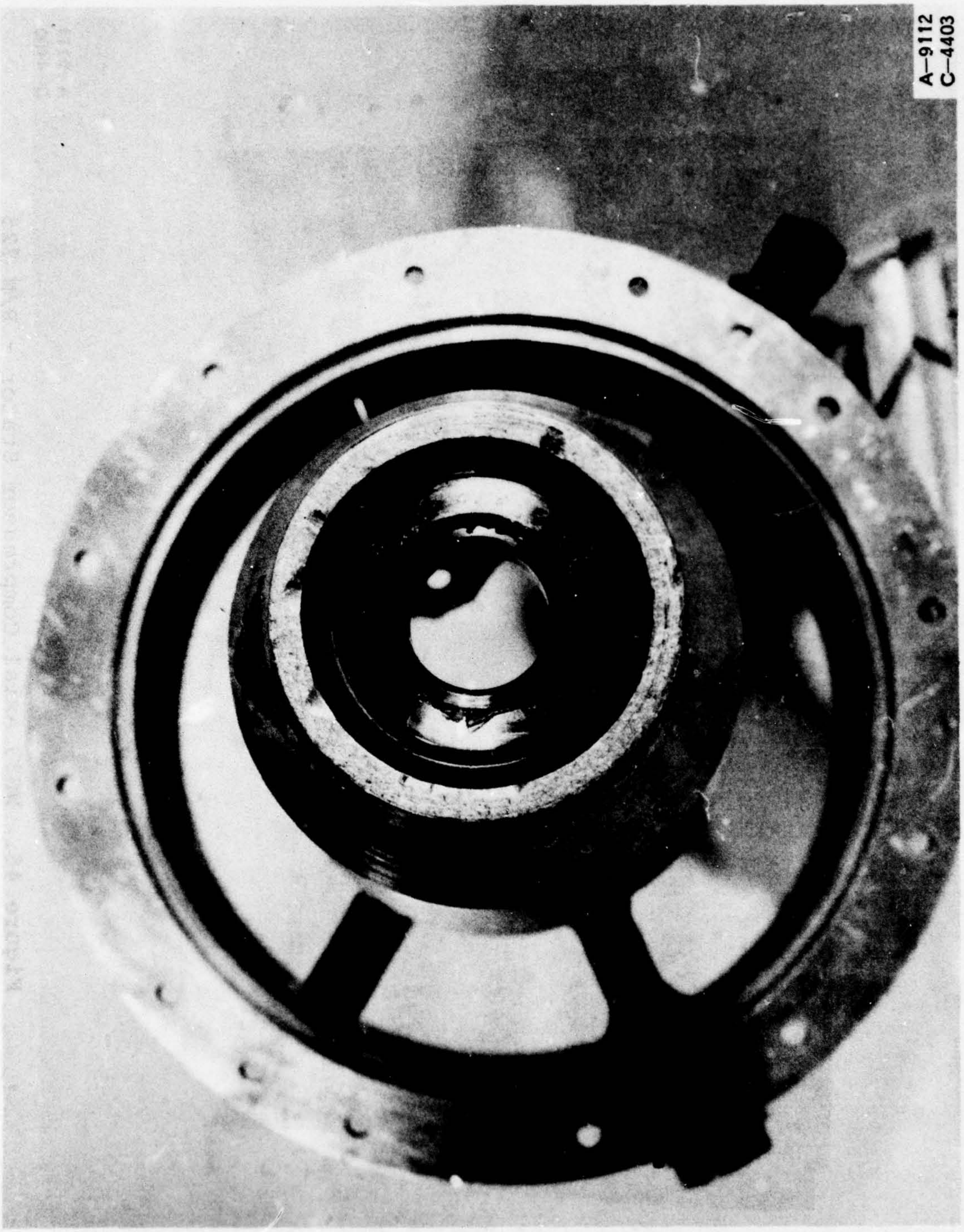
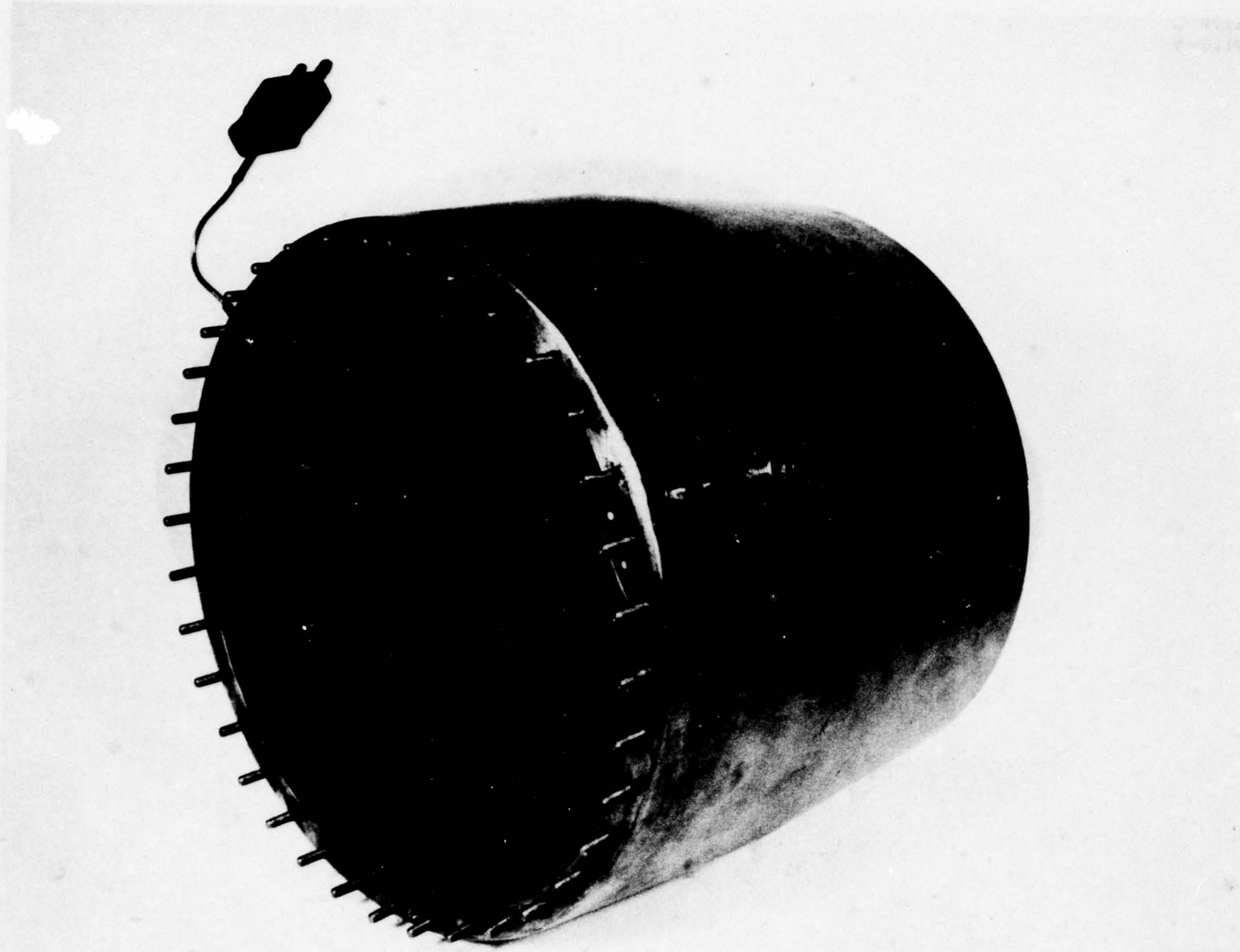
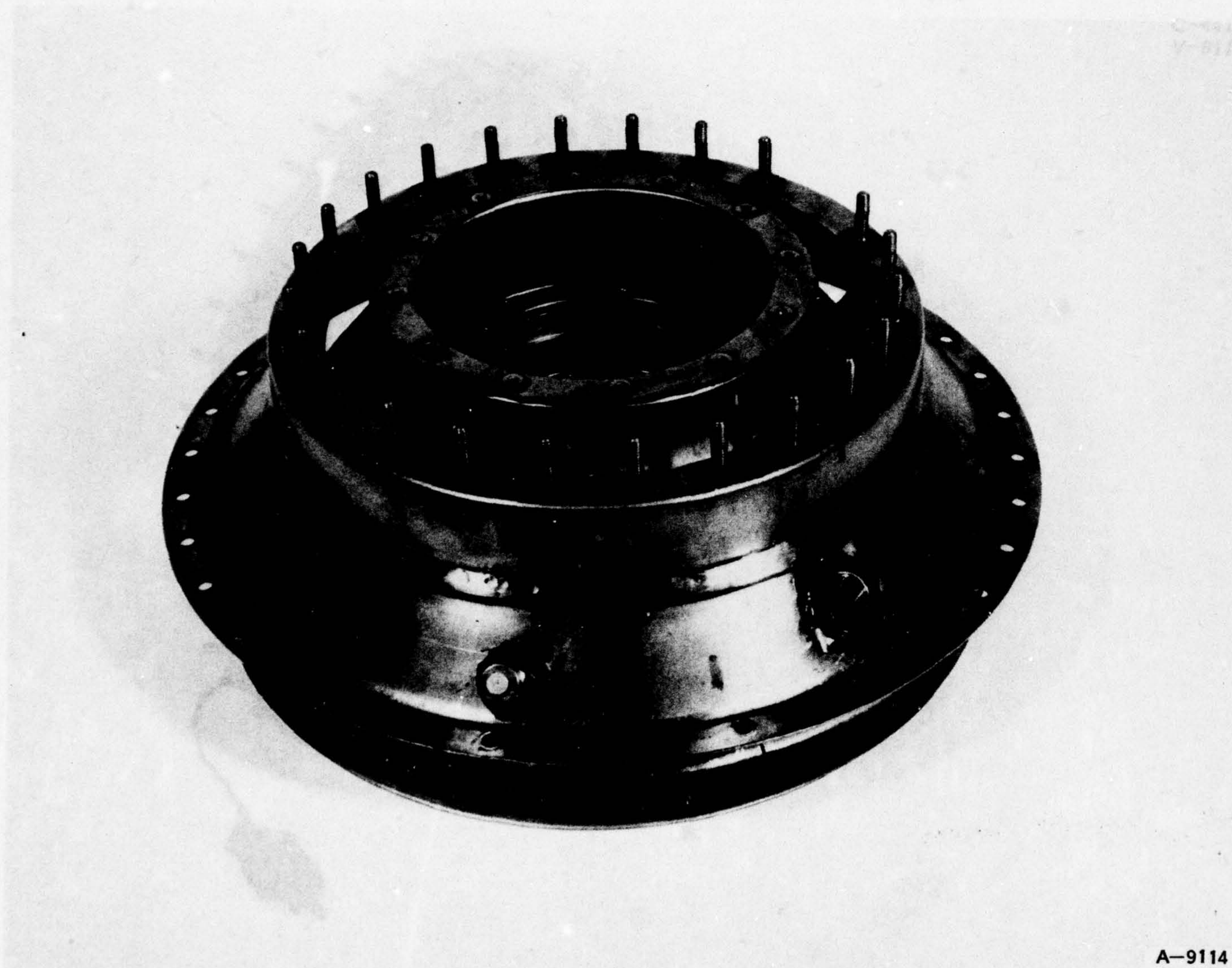


Figure 47. WR33 Inlet Housing - P/N 750



A–9113
C–4419

Figure 48. WR33 Main Housing - P/N 600



A–9114
C–4413

Figure 49. WR33 Diffuser and Combustor Cover - P/N 650 (View 1)

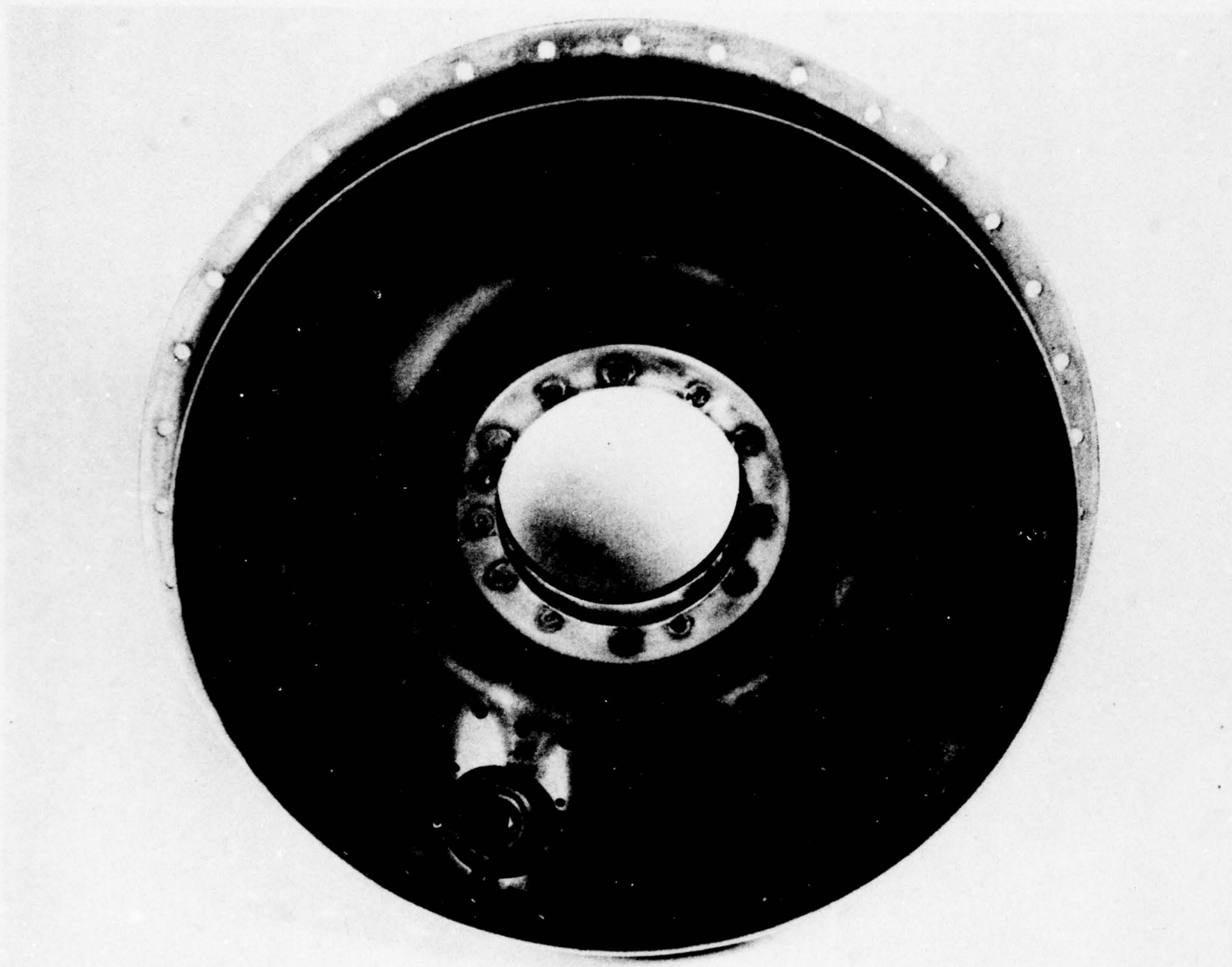


Figure 50. WR33 Diffuser and Combustor Cover - P/N 650 (View 2)

APPENDIX A

MANUFACTURING PARTS LIST

WR33 LOW COST EXPENDABLE GAS TURBINE ENGINE PARTS LIST

1. 100 Combustor Assembly
2. 200 Turbine Shaft Assembly (230 option)
3. 300 Turbine Shroud Assembly
4. 400 Tailcone and Rear Bearing Support Assembly
5. 450 Rear Bearing Housing Assembly
6. 600 Combustor Case Assembly
   613 Assy - Clamp Ring
7. 650 Diffuser and Combustor Cover Assembly
8. 700 Fuel Transfer and Shaft Seal Housing Assembly
9. 720 Assy - Axial Compressor Rotor (with manifold)
   722 Axial Compressor Rotor (6 Stage) Casting (719 option integral)
   723 Axial Compressor Stator (6 Stage) Mach.
10. 725 Axial Compressor Stator (6 Stage) Mach.
    726 Axial Compressor Stator (6 Stage) Casting
11. 750 Compressor Inlet Housing
12. 850 Nose Cone Assembly
13. 900 Std Parts List
14. 950 Assy Dwg

P/N	ITEM	TITLE	REQ'D
400		Tailcone, Rear Brg Support Assembly	1
401	1	Wall-Outer	1
403	2	Wall-Inner	1
405	3	Flange-Support	1
117	4	Weld Stud	2
404	5	Flange-Heat Shield	2
117	6	Weld Stud	4
402	7	Vane	3
502	8	Tailplug	1
500	9	Strap Assembly	1
501	10	Retaining Strap	1
117	11	Weld Stud	1
122		Locknut-Tailplug	1
450		Rear Bearing Housing Assembly	1
451	1	Housing	1
452	2	Airshield	1
125	5	Dowel	2
412		Snap Ring	1
453	4	Cover-Heat Shield	2
122		Locknut	4
406	12	Insulation	Var.

P/N	ITEM	TITLE	REQ'D
200		Turbine Shaft Assembly	1 230 Option
205	6	Hub-Nozzle Mounting	1
212	11	Main Shaft	1
213	12	Turbine Cast	1 19-9 Cstg 31022
214	13	Turbine Mach.	1
215	14	Braze Material - Nickel	UAR
217	16	Shaft - Forward	1
411		Bearing - Rear	1
705		Bearing Assy Thrust (27041)	1
710		Bearing (22415)	1
711		Shield (27045)	2
715		Shield - Rear Bearing	2
712		Grease (27042)	UAR
707		Locknut - Thrust Bearing	1
708		O-Ring - Seal Runner	1
709		Seal Runner - Fuel (24046)	1
209		Fuel Nozzle	4
713		Drive Pin	2
714		Spacer	1
300		Turbine Shroud Assembly	1
301	1	Flange	1

P/N	ITEM	TITLE	REQ'D
302	3	Shroud	1
100		Combustor Assembly	1
101	1	Outer Wall	1
108	2	Inner Liner Support	1
114	3	Rivet	2
115	4	Liner-Inner	1
116	5	Strip Joint	1
107	6	Shroud Shaft	1
105	7	Inner Wall	1
111	8	Seal	2
110	9	Seal	2
114	10	Rivet	4
109	11	Support Balance Seal	1
113	12	Seal	2
112	13	Seal	2
106	14	Support-Shaft Shroud	1
103	15	Vane Pressure Side	15
104	16	Vane Suction Side	15
102	19	Flange-Mounting	1
613		Assy-Clamp Ring	1
117	20	Weld Stud	30
611		Ring	1
612		Bracket	3

P/N	ITEM	TITLE	REQ'D
120	21	Strip Joint	1
118	22	Liner-Outer 2.70 wide x 27.48 long	1
119	23	Strip Joint 2.70 long x .60 wide	1
150	24	Flange - Air Start	1
151	25	Tube - Air Start	1
650		Diffuser and Combustor Cover Assembly	1
651	1	Cover	1
652	2	Support	6
653	3	Outer Wall-Diffuser	1
117	4	Weld Stud	24
655	5	Inner Wall-Diffuser	1
114	6	Rivet	12
657	7	Support	1
658	8	Cover-Burner	1
659	9	Lab Seal	1
660	10	Lab Seal	1
661	11	Igniter Boss	1
149		Igniter	1
605	12	Locktite 290	UAR
600		Combustor Case Assy	1
606		Case	1
607		Rivet	3
610		Ring Assy	1

P/N	ITEM	TITLE	REQ'D
608		Ring	1
117		Weld Stud	36
609		Bracket	3
725		Compressor Stator Assembly	1
726		Housing - Cast	1
		- Vanes-726-401 thru 726-406	-X-
728		Shim	2
125		Dowel	8
126		Washer	72
127		Screw	24
122		Locknut	48
719		Comp Rotor (Integral Cost)	(719 Option)
722		Compressor Rotor - Cast	1
723		Compressor Rotor - Mach	1
724		Washerplate	1
137		Screw - 1/4 - 28 x 2.375 long	4
141		Body 32067	1
142		Poppet 32068	1
143		Spring 15611	1
144		Seat 15587	1

P/N	ITEM	TITLE	REQ'D
145		Pin MS171506 (191178)	1
146		Lubricant 17734	UAR
147		Primer 15686	UAR
148		Adhesive 15688	UAR
850		Nose Cone Assembly	1
851		Nose Cone	1
852		Retaining Strap	2
853		Pin	2
			M/F 117
700		Fuel Transfer and Seal Housing Assembly	1
701	1	Housing	1
702	2	Fitting - Air	1
133	3	Seal - Carbon	1
134	4	O-Ring - Seal	1
135	5	O-Ring - Housing	3
137		Screw - 1/4 - 28 x 2.375 long	3
138		Washer	3
750		Compressor Inlet Housing (Fab)	1
751		Inlet	1
752		Bearing Support	1
130		O-Ring	1
129		Union-Fuel - 6	1
131		O-Ring	1
132		Union-Air - 3	1
753		Snap Ring	1

STANDARD PARTS SUMMARY

	P/N	QUAN
Locknuts - MS21043-08	122	119
Dowels - Groove Pin Corp., 50-125-0375 or MS171524 (91100)	125	8
Washers - MS9321-08 (.05 OT x .375 OD) CRS	126	51
Weld Studs - Gripco (Triwest), UN-0832-x-0625	117	100
Rivets - USM (Triwest), SSD42SSBS	114	22
WRC P/N 90051	607	3
Bearing Locknut - Nylok Detroit, BR-N-03-1 - Torkon, TBL-N03	707	1
Fuel Nozzle - Steinen, 5 GPH, Type PH, 30°	209	4
Screws - Allen		
#8-32 x .75 long	127	24
1/4 - 28 x 2.375 long	137	7
Sparkplug - R6Y	149	1
Carbon Seal - Sealol (WRC P/N 24045)	133	1
Seal Runner (WRC P/N 24046)	709	1
Snaprings - Spirolox, RR-137S	412	1
Truarc-5008-156-pp (91047)	753	1
O-Rings - Precision, -026 size Viton	134	1
-039 size Viton	135	2
-012 size Viton	708	1
Bearings - 203 Barden S203SST15-20G29	705	1
103 Barden 103SST5G18	411	1
O-Ring - 903 size Viton	131	1
-3 Union, Stl or Alum	132	1
O-Ring - 906 size Viton	130	1
-6 Union, Stl or Alum	129	1
Locktite 290	605	UAR

APPENDIX B

ENGINE DATA REDUCTION

The approach used to calculate engine performance characteristics for the Mach number simulated endurance run is outlined in the following paragraphs.

The endurance run test data was initially run through the standard data reduction program. The resultant reduced data is completely valid for the Mach number simulated run except for the values of net thrust (Fn) and specific fuel consumption (SFC). A meaningful thrust measurement could not be made since the exhaust diffuser used to reduce the nozzle exit static pressure was attached to the engine. Therefore, gross thrust was calculated from measured values of total pressure, static pressure and total temperature at the nozzle exit. The calculation sheet used to determine Fn and SFC is shown in Table B-I. Data points 5 through 12 were obtained with the exhaust diffuser in place. Data points 13, 14 and 15 were obtained without the exhaust diffuser in place where a valid measurement of Fn could be obtained. These three data points therefore, served as calibration points for the calculation of thrust from the measured nozzle total pressure and temperature as explained later. The calculation procedure used is as follows:

1. From the calculated value of Ps exit/P_{t1} - (Step 9) the simulated value of flight Mach number Mn - (10) and T_{s1}/T_{t1} - (11) were determined from gas tables.

2. The simulated value of Tamb (12) was obtained from the following equation. $Tamb = (T_S/T_{T1})*T_{T1}$.

TABLE B-I. ENGINE DATA REDUCTION

STEP	DATA PT. #	5	6	7	8	9	10	11	12	13	14	15
1	$^{*}P_{T_I}$ -psia	14.349	14.349	14.349	14.349	14.349	14.349	14.349	14.349	14.349	14.349	14.349
2	$^{*}T_{T_I}$ - ^{o}R	486.	485.	485.5	484.5	485.	485.	485.	484.	494.	493.	492.
3	$^{**}S_{T_I}$	0.9764	0.9764	0.9764	0.9764	0.7964	0.9764	0.9764	0.9764	0.9764	0.9764	0.9764
4	$^{**}\theta_{T_I}$	0.9369	0.9350	0.9360	0.9341	0.9350	0.9350	0.9350	0.9331	0.9524	0.9505	0.9485
5	$^{*}P_{T_{noz}}$ "H_{gG}	20.23	20.17	20.2	20.17	20.27	20.27	20.2	13.03	9.8	15.57	20.33
6	$P_{T_{noz}}$ -psia	24.28	24.26	24.27	24.26	24.30	24.30	24.27	20.75	19.16	21.99	24.33
7	$^{*}P_{S_{exit}}$ "H_{gG}	- 9.4	- 9.45	- 9.8	- 9.8	- 9.8	- 9.8	- 9.75	- 9.6	0.	0.	0.
8	$P_{S_{exit}}$ (pamb)-psia	9.733	9.709	9.537	9.537	9.537	9.537	9.537	10.126	14.349	14.349	14.349
9	$P_{S_{exit}}/P_{T_I}$	0.6783	0.6766	0.6646	0.6646	0.6646	0.6646	0.6646	0.7057	1.0	1.0	1.0
10	Mn	0.767	0.769	0.788	0.788	0.788	0.788	0.788	0.725	0.	0.	0.
11	$(T_S K_T)_1$	0.8958	0.8953	0.8907	0.8907	0.8907	0.8907	0.8907	0.9059	1.0	1.0	1.0
12	T_{amb} ^{o}R	435.3	434.2	432.4	431.5	431.9	431.9	431.9	438.4	494.	494.	494.
13	$^{**}N_1/\sqrt{\theta_{T2}}$%	100.6	100.7	100.8	100.8	100.9	100.8	100.9	95.1	85	94	99.9
14	$P_{S_{exit}} P_{T_{noz}}$	0.4007	0.4003	0.3929	0.3937	0.3924	0.3924	0.3929	0.4596	0.7488	0.6573	0.589
15	$M_{noz\ exit}$	1.0	1.0	1.0	1.0	1.0	1.0	1.0	1.0	0.671	0.823	0.921
16	$(V/\sqrt{TT})_{noz}$	44.272	44.272	44.272	44.272	44.272	44.272	44.272	44.272	30.94	37.30	41.22
17	$(P_{S_{noz}} - P_{amb})$	3.387	3.395	3.575	3.567	3.593	3.593	3.575	1.672	0.0	0.0	0.0
18	$^{*}EGT$ ^{o}R	1806	1808	1801	1803	1800	1796	1798	1551	1616	1720	1851
19	V_{jet} - ft/sec	1881.1	1882.5	1878.8	1879.9	1878.3	1876.2	1877.3	1743.6	1243.6	1546.9	1773.4
20	$^{**}W_A$ - lb/sec	3.701	3.705	3.707	3.716	3.709	3.707	3.714	3.427	2.843	3.309	3.633
21	$F_{gross\ ideal}$	264.6	265.2	267.4	267.9	267.7	267.4	267.6	209.5	109.9	159.1	200.2
22	$^{**}F_{Gmeas}$	-----	-----	-----	-----	-----	-----	-----	-----	106.2	147.4	186.0
23	C_F	0.93	0.93	0.93	0.93	0.93	0.93	0.93	0.93	0.9667	0.9265	0.9292
24	$F_{gross\ rlal}$	246.0	246.6	248.7	249.1	248.9	248.7	248.8	194.8	-----	-----	-----
25	V_O	783.2	784.2	801.8	800.9	801.4	801.4	801.4	742.9	-----	-----	-----
26	F_{ram}	90.1	90.3	92.4	92.5	92.4	92.3	92.5	89.1	-----	-----	-----
27	F_{net}	155.9	156.3	156.3	156.6	156.5	156.4	156.2	115.7	-----	-----	-----
28	$^{**}W_F$	253.8	252.4	251.6	251.3	251.3	251.1	251.0	188.6	-----	-----	-----
29	SFC	1.628	1.615	1.609	1.605	1.606	1.605	1.607	1.630	-----	-----	-----

* From engine test data

** From data reduction program

A-9245

3. The nozzle exit Mach number $M_{noz\ exit}$ (15) was determined with gas tables from the calculated value of $P_{S_{exit}}/P_{T_{noz}}$ -(14). The nozzle pressure ratio over critical for data points 5 through 12 therefore the nozzle exit Mach number was set at 1.0 for these points.

4. Values of $V/\sqrt{T}_{T_{noz}}$ -(16) were obtained with gas tables from the value of $M_{noz\ exit}$ -(15).

5. The value of $(P_{S_{noz}} - P_{amb})$ -(17) was obtained for the chocked nozzle data points from the following equation.

$$(P_{S_{noz}} - P_{amb}) = (P_S/P_T)^* * P_{T_{noz}} - P_{S_{exit}}$$

For data points 13, 14 and 15 where the nozzle is unchoked the value is zero since $P_{S_{noz}} = P_{amb}$.

6. The jet nozzle exit velocity V_{JET} - (19) was obtained from:

$$V_{JET} = (V/\sqrt{T}_T)_{noz} * \sqrt{EGT}.$$

7. The ideal gross thrust (21) was now calculated from the following equation:

$$F_{gross\ ideal} = (Wa\ V_{jet})/g_o + (P_{S_{noz}} - P_{amb})A_{jet}.$$

8. The value of CF(23) was now calculated for data points 13, 14, and 15 where a valid measurement of thrust could be made. From these results a value of CF of 0.93 was selected for data points 5 through 12.

9. The real gross thrust (24) was calculated for data points 5 through 12 with the following equation:

$$F_{gross\ real} = F_{gross\ ideal} * CF$$

10. Flight velocity (25) was now calculated:

$$Vo = Mn * \sqrt{\gamma g R T_{amb}}$$

11. Ram drag (26) was calculated from the following equation:

$$F_{RAM} = (Wa * Vo)/g_o$$

12. Net thrust (27) was now calculated:

$$F_{net} = F_{gross\ real} - F_{ram}$$

13. Specific fuel consumption (29) was calculated from the following equation:

$$SFC = W_F/F_{net}$$

From the above data it is also possible to estimate engine performance at the sea level, Mach number 0.7

standard day condition for the maximum engine rating. At this condition the engine inlet temperature is $569.5^{o}R$ therefore:

$$\theta_{TI} = 1.09805$$

$$N/\sqrt{\theta_{TI}} = 100\%/\sqrt{1.09805}$$
$$= 95.5\%$$

The engine was run at approximately the above corrected speed during data point 12. Therefore the following additional corrections to this data point will give the engine thrust and SFC levels which would occur at sea level, Mach .7 standard day conditions.

Data Point 12:

Fn	115.7 lb
SFC	1.630 lb/(lb-hr)
S_{amb}	0.6890
θ_{amb}	167.9 lb
$SFC/\sqrt{\theta_{amb}}$	1.773 lb/lb-hr)

The above thrust and SFC levels are from Engine S/N 2. Engine S/N 1 levels were obtained by ratioing the SLS levels of performance of the two engines at an engine corrected speed of 95.4 percent rpm by the following method:

Engine	S/N 1	S/N 2
$N/\sqrt{\theta}$	95.4%	95.4%
Fn	166.06	160.55
SFC	1.389	1.429

Therefore:

$$(Fn)_{ENG\ 1\ MN\ 0.7} = \frac{166.06}{160.55}\ (167.9)$$

$$= 173.7\ lb$$

$$(SFC)_{ENG\ 1\ Mn\ 0.7} = \frac{1.389}{1.429}\ (1.773)$$

$$= 1.724\ lb/lb\text{-}hr)$$

APPENDIX C

ABBREVIATIONS/ACRONYMS USED IN THIS REPORT

AFAPL	Air Force Aero Propulsion Laboratory
AMB	Ambient
ASME	American Society of Mechanical Engineers
ASSY	Assembly
BRG	Bearing
CY	Calendar Year
DC$	Direct Charge Dollars
DWG	Drawing
EGT	Exhaust Gas Temperature
Fn	Net Thrust
Fy	Fiscal Year
G&A	General and Administrative
HSG	Housing
ICN	Interface Control Notice
ID	Inside Diameter
Mach	Mach Number
MATL	Material
NOZ	Nozzle
P/N	Part Number
psig	Pounds-force per Square Inch Gage

R&D	Research and Development
RFP	Request for Proposal
rpm	Revolutions per Minute
SFC	Specific Fuel Consumption
SL	Sea Level
S/N	Serial Number
TIT	Turbine Inlet Temperature
TP	Tail Pipe
USAF	United States Air Force
WRC	Williams Research Corporation

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